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ROYAL AERONAUTICAL SOCIETY 1866

Journal

of

THE ROYAL AERONAUTICAL SOCIETY

AUGUST 1956

CONTENTS

NOTICES

THE FIRST CHADWICK MEMORIAL LECTURE
THE LIFE AND WORK OF ROY CHADWICK — H. ROGERSON

PROPELLERS FOR MILITARY AND CIVIL AIRCRAFT
L. G. FAIRHURST

THE AERODYNAMICS OF THE HELICOPTER IN FORWARD FLIGHT — E. N. BRAILSFORD

TECHNICAL NOTES

A Note on the China-Film Technique for Boundary Layer Indication—R. J. STALKER. Further Note on the Treatment of Structural Discontinuities —W. J. GOODEY. The Estimation of the Lift due to Full and Part Span Controls—J. C. SIMMONS. and J. A. DUNSBY. Helicopter Stability in Hovering Flight—Comments by A. R. S. BRAMWELL, L. GOLAND, K. HOHENEMSER. Reply by P. R. PAYNE. Power Loss and Fuel Pressure Rise due to Centrifugal Pumping in Tip-Driven Helicopter Rotors—Comments by A. CZERNIN. Reply by P. R. PAYNE.

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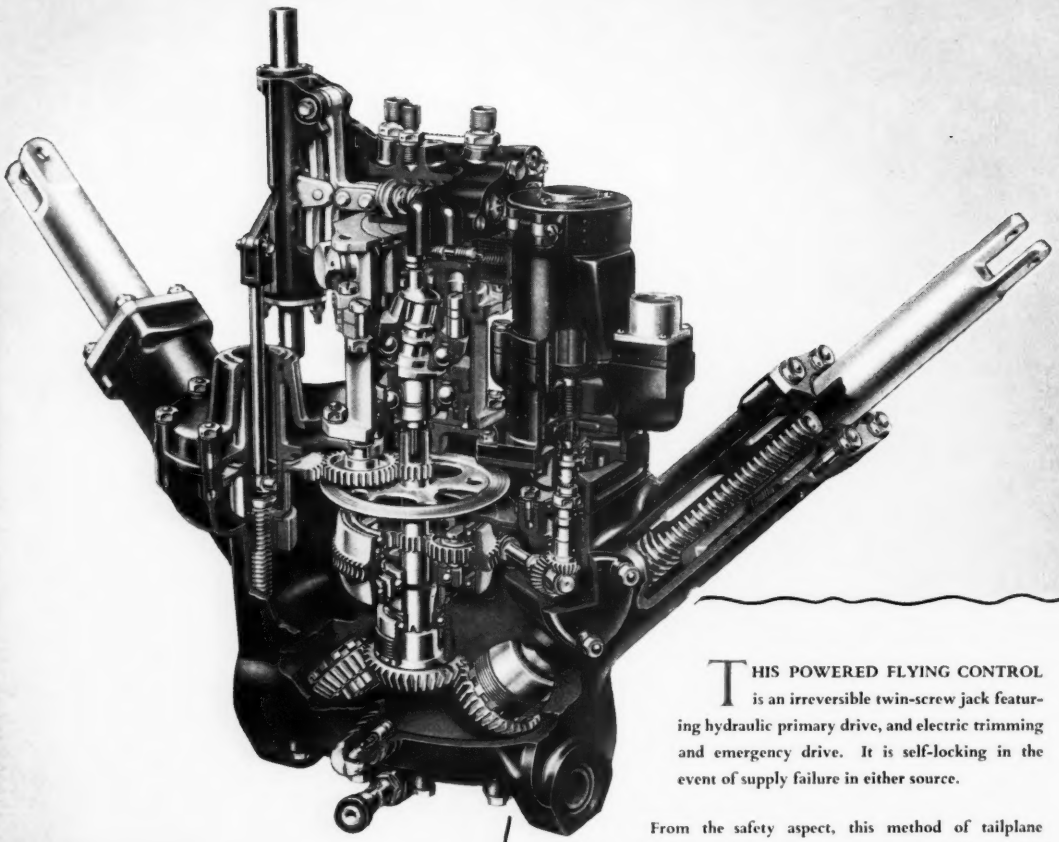

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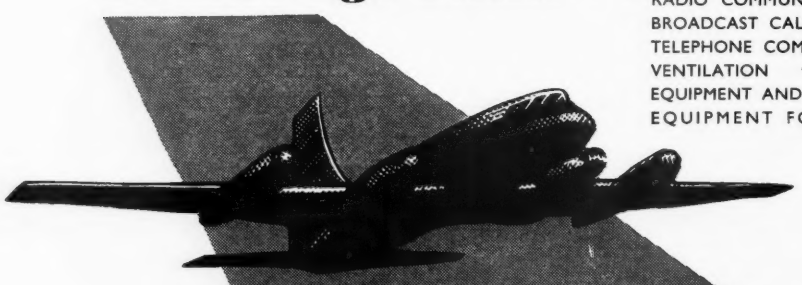

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The issue has been prepared by the Royal Aeronautical Society in conjunction with the Institute of Petroleum.

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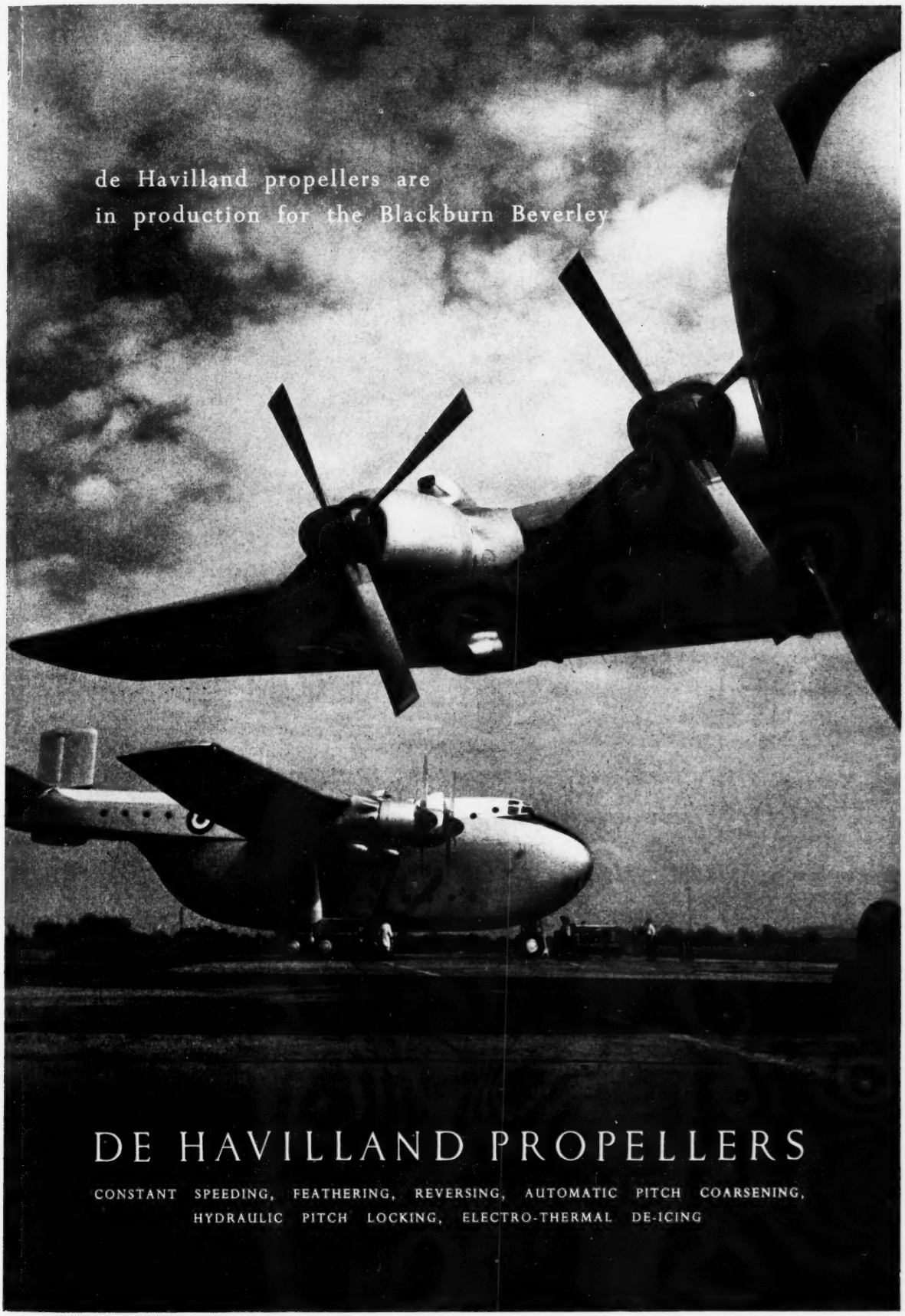
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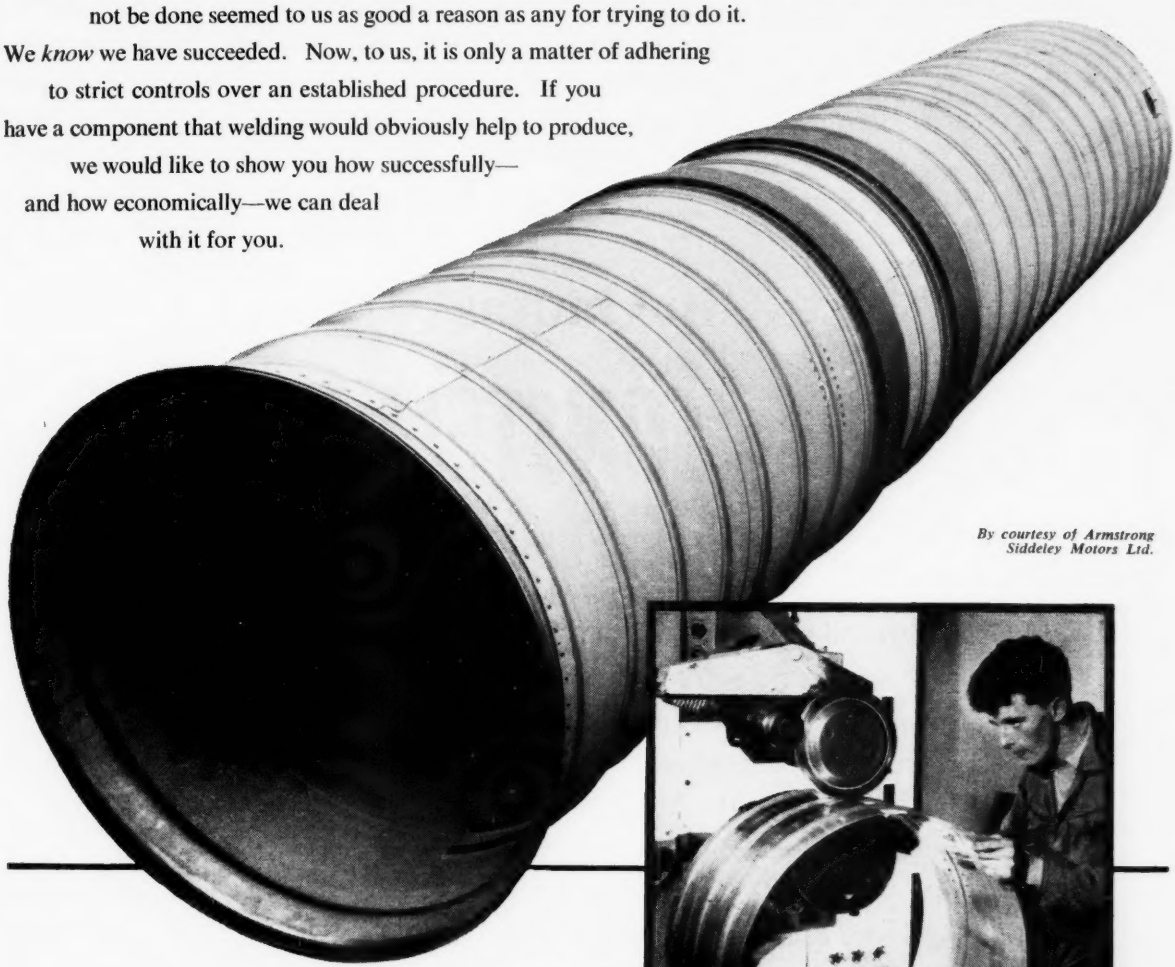

By courtesy of Armstrong Siddeley Motors Ltd.

Photograph on the right, shows a section of the complete Shroud (above) in D.T.D. 610 for the Sapphire Jet Engine being spot-welded on one of the Taylor, Winfield Roll-Spot welding machines installed at "Middleton." More than 10,000 Spots are made on each engine set.

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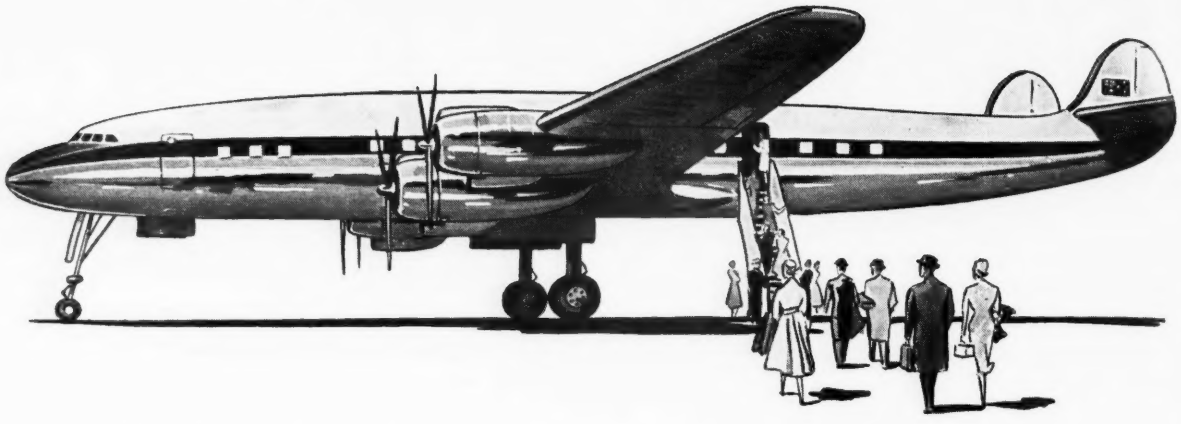

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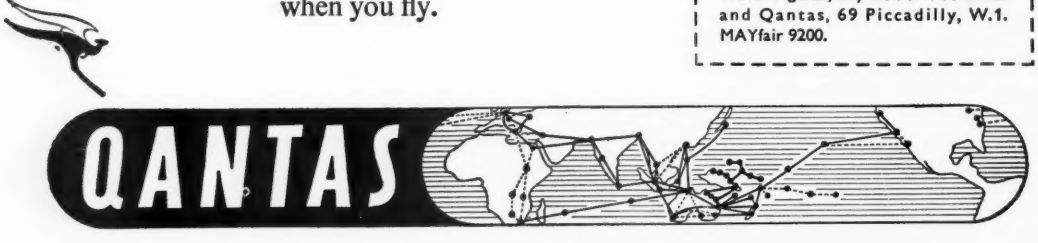

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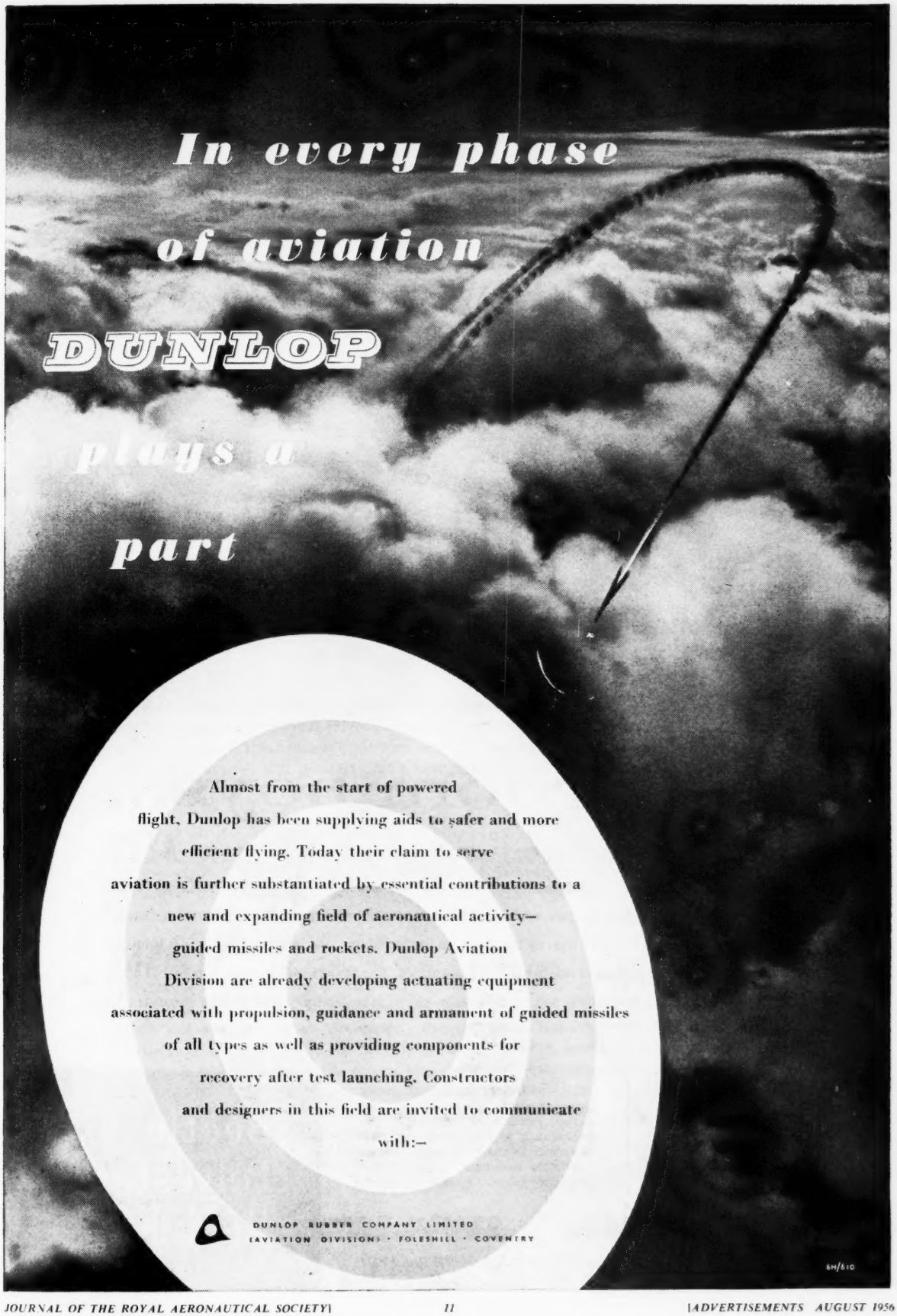
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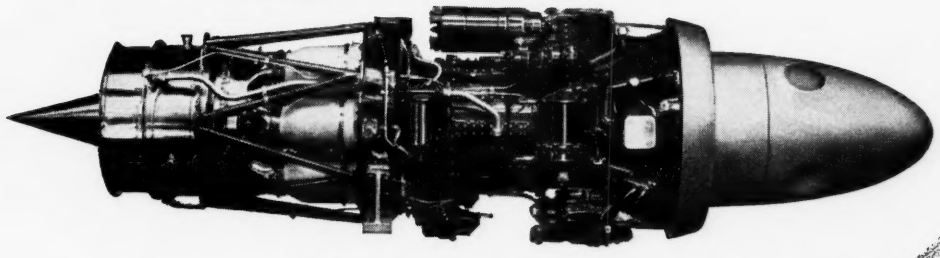
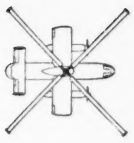
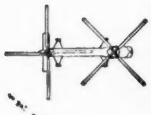




ELAND The Convair 340 aircraft converted to Eland engines will be on demonstration flights in Europe during this year. Another aircraft conversion to *Elands* is the Elizabethan, which will be evaluated on BEA routes. Napier *Elands* fitted with auxiliary compressors for the tip-driven rotor system, will power Fairey's new large transport helicopter, the Rotodyne.

ORYX The Napier *Oryx* 780-950 gas h.p. turbo-gas-generator provides the hot gas which is ducted to the rotor head for the propulsion of helicopters by jet reaction at the rotor blade tips. This system eliminates all mechanical transmission. The *Oryx* has been officially Type Tested at 780 and 865 gas h.p.



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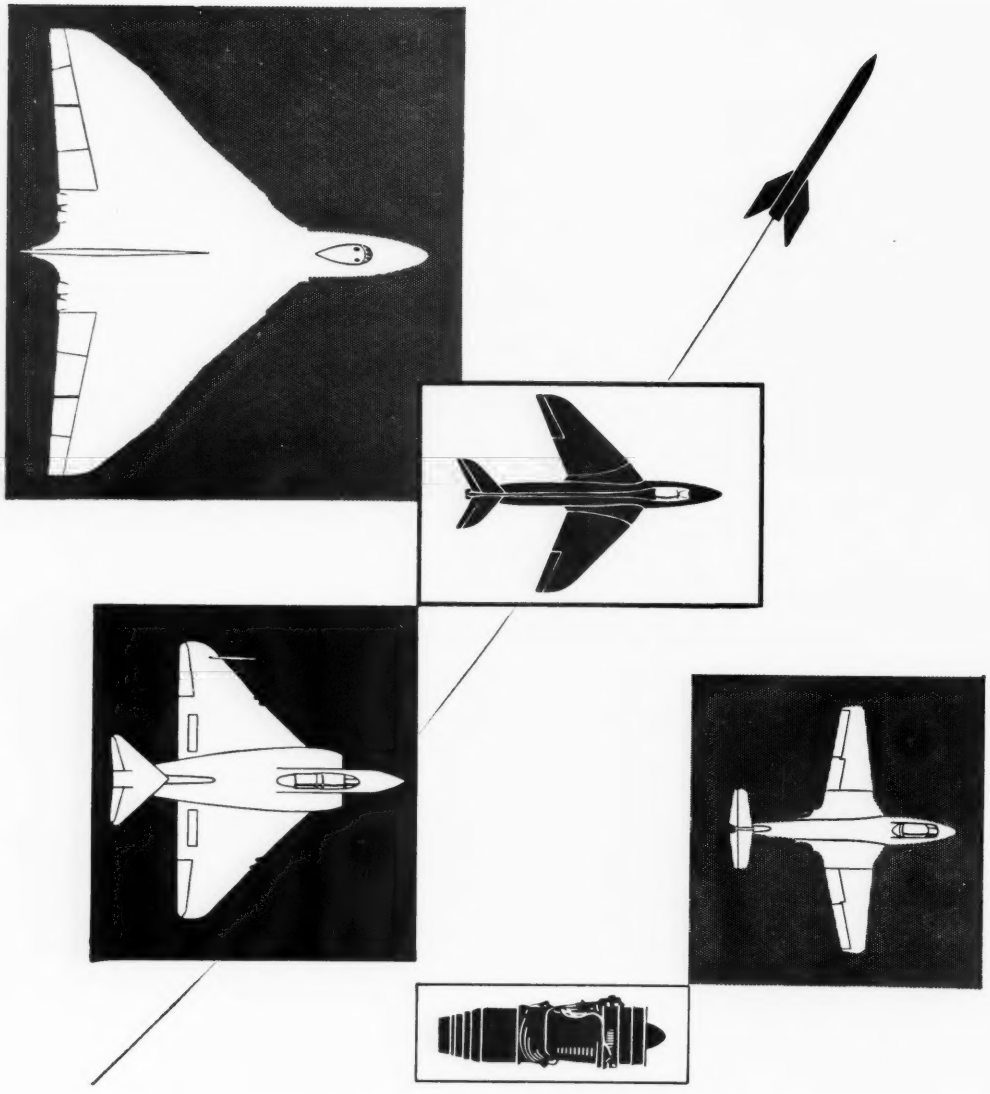

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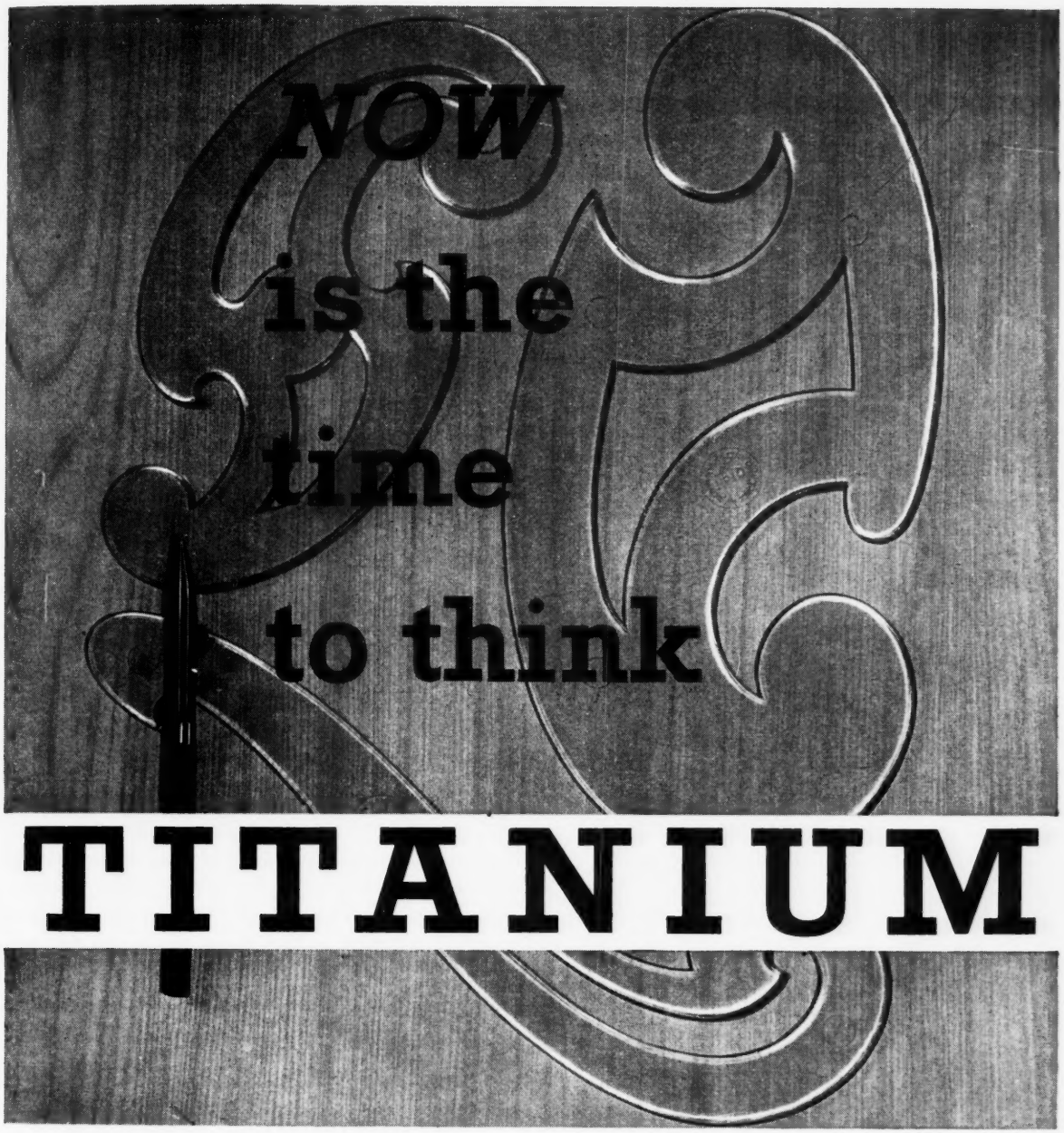

TITANIUM

TALKING POINTS No. 1 . . . Titanium and Corrosion-Fatigue

Nowadays every engineer is familiar with the phenomenon of metal fatigue. Recognising that any piece of metal will fail by alternations or repetitions of a stress far lower than that needed to break it in a single application, he fixes his allowable stresses accordingly.

The fatigue strength of metals is, however, reduced still further if the alternating or repeating stresses are accompanied by even mildly corrosive conditions. With most metals, allowable stresses must be substantially reduced to prevent failure by corrosion-fatigue.

Titanium is an exception. Tests carried out in air and in seawater show that the fatigue resistance of titanium is not affected at all by the presence of seawater—a result not found with the very large number of metals similarly tested.

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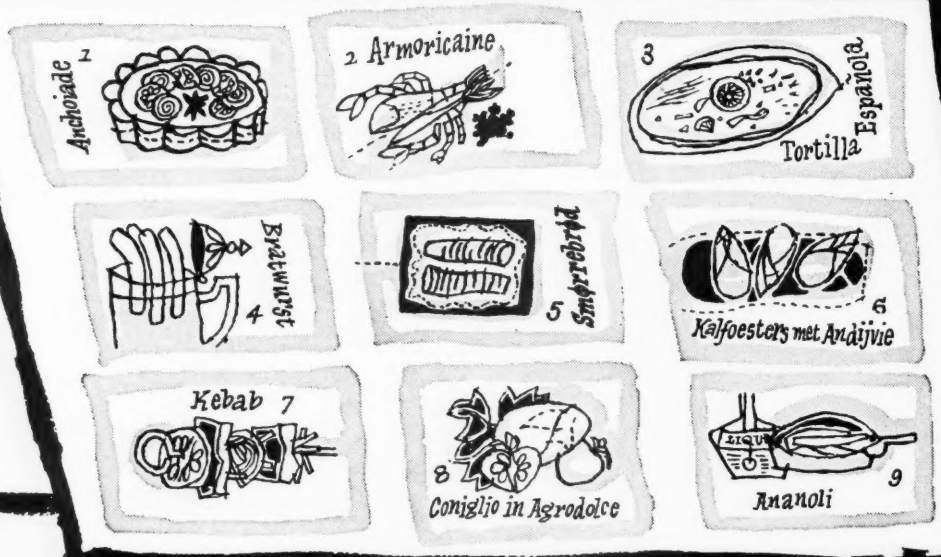
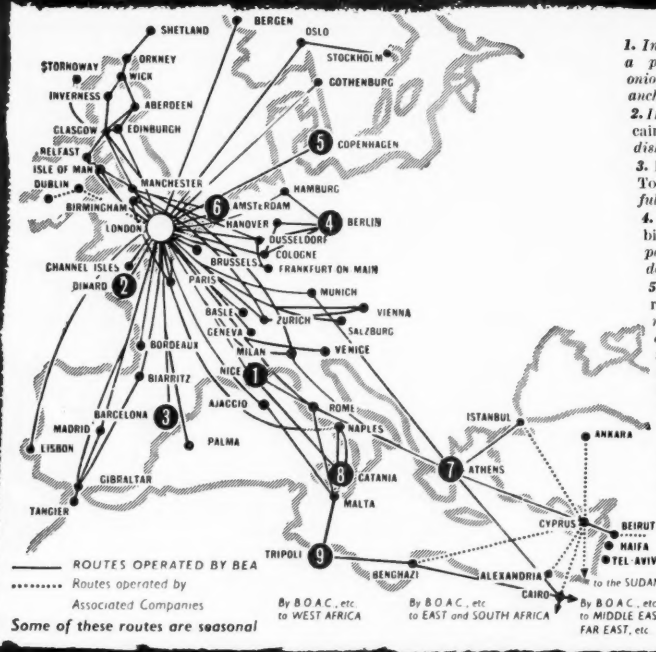
the BEA gastronomical map of Europe





1. *In* Nice *try* Anchoiade: *a pastry with tomatoes, onion and fresh salty anchovies.*

2. *In* Dinard *go for* Armoricaine — *a lovely hot lobster dish.*

3. Barcelona's *the place for* Tortilla Española, *a colourful Spanish-style omelette.*

4. *Don't miss* Bratwurst in bier nach Berliner art: *pork sausages in beer — done the* Berlin *way.*

5. *In* Copenhagen, Smørrebrød *is a "must" — marvellous filleted smoked eel and scrambled egg on rye bread.*

6. *In* Amsterdam, Kalfoesters met Andijvie *(fried veal steaks with endive) are delicious.*

7. *In* Athens *try* Kebab — *mutton with chopped onions, bay leaves and tomatoes grilled on a skewer.*

8. *A great favourite in* Sicily *is* Coniglio in Agrodolce: *rabbit in sweet and sour sauce.*

9. Ananoli — *banana fritters soaked in liqueur will delight you in* Tripoli.

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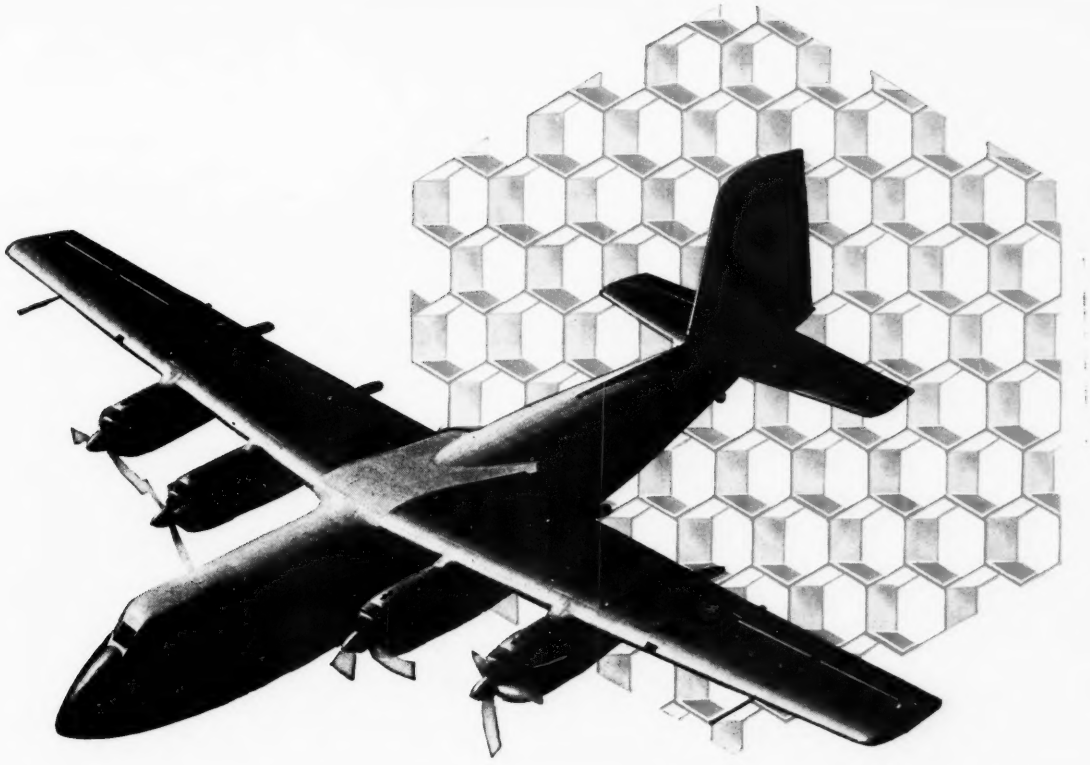
Honeycomb



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VOLUME 60 AUGUST 1956 NUMBER 548

CONTENTS

PAGE

Notices XLVII

Canadian Aeronautical Institute—Associate Fellowship Examinations, December 1956—News of Members—Institute of Physics: Joint Conference—Elections—Graduates' and Students' Section Visits—Aviation in Russia 1911—Anglo-American Conference Proceedings 1955—Changes of Address or Appointment—Journal Binding

The First Chadwick Memorial Lecture
The Life and Work of Roy Chadwick — H. ROGERSON — 501

Propellers for Military and Civil Aircraft — L. G. FAIRHURST — 515

The Aerodynamics of the Helicopter in Forward Flight
E. N. BRAILSFORD — 523

Technical Notes 543

A Note on the China-Film Technique for Boundary Layer Indication—R. J. STALKER. Further Note on the Treatment of Structural Discontinuities—W. J. GOODEY. The Estimation of the Lift due to Full and Part Span Controls—J. C. SIMMONS and J. A. DUNSBY. Helicopter Stability in Hovering Flight.—Comments by A. R. S. BRAMWELL, L. GOLAND, K. HOHENEMSER. Reply by P. R. PAYNE. Power Loss and Fuel Pressure Rise due to Centrifugal Pumping in Tip-Driven Helicopter Rotors.—Comments by A. CZERNIN. Reply by P. R. PAYNE.

Graduates' and Students' Section 556

The Library 557

REVIEWS: Creep and Fracture of Metals at High Temperatures—Instrument Engineering, Volume III; Applications of the Instrument Engineering Method; Part I, Measurement Systems—Rockets and Guided Missiles—Let's Learn to Fly—The Science and Practice of Welding—Strength of Materials.

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Hon. Secretary: R. W. HILKEN, D.F.C., A.F.R.Ae.S., Blackburn & General Aircraft Ltd., Brough.

CHESTER
President: J. C. CORBY, O.B.E.
Chairman: A. W. TURNER.
Hon. Secretary: B. V. POULSTON, Thornton Research Centre, P.O. Box 1, Chester.

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President: W. A. TAMBLIN, F.R.Ae.S.
Chairman: G. B. EVANS.
Hon. Secretary: H. E. VINCENT, A.F.R.Ae.S., de Havilland Aircraft Co. Ltd., Airspeed Div., Christchurch Aerodrome, Hants.

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Chairman: W. J. PETERS.
Hon. Secretary: C. T. SCULTHORPE, A.F.R.Ae.S., Sir W. G. Armstrong Whitworth Aircraft Ltd., Baginton, Coventry.

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Chairman: A. G. ELLIOTT, C.B.E., F.R.Ae.S.
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President: Professor W. J. DUNCAN, C.B.E., F.R.Ae.S.
Chairman: D. B. SWINBANKS, A.F.R.Ae.S.
Hon. Secretary: Dr. A. W. BABISTER, A.F.R.Ae.S., Dept. of Aeronautics, The University, Glasgow, W.2.

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President: R. H. BOUND, F.R.Ae.S.
Chairman: J. F. CUSS, A.F.R.Ae.S.
Hon. Secretary: R. C. DELL, A.R.Ae.S., Dowty Equipment Ltd., Arle Court, Cheltenham.

HALTON
Chairman: Gp. Capt. A. K. HUNTER, O.B.E., A.F.R.Ae.S.
Hon. Secretary: Flt. Lt. B. D. AUSTEN, A.F.R.Ae.S., Apprentice Schools, R.A.F., Halton.

HATFIELD
President: Sir GEOFFREY DE HAVILLAND, C.B.E., A.F.C., [Hon.F.R.Ae.S.
Chairman: C. T. WILKINS, O.B.E., F.R.Ae.S.
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President: Professor A. J. MURPHY, A.F.R.Ae.S.
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Hon. Secretary: Sqn. Ldr. E. J. HOLDEN, R.A.F. Technical College Mess, Henlow, Bedfordshire.

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Chairman: H. KNOWLER, F.R.Ae.S.
Hon. Secretary: L. W. ROSENTHAL, A.F.R.Ae.S., Saunders-Roe Ltd., Osborne, E. Cowes, Isle of Wight.

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Chairman: I. TORBE, A.F.R.Ae.S.
Hon. Secretary: R. G. AUSTIN, A.R.Ae.S., Helicopter Dept., Auster Aircraft, Rearsby, Leicester.

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President: W. A. SUMMERS, A.F.R.Ae.S.
Chairman: E. N. B. BENTLEY, A.F.R.Ae.S.
Hon. Secretary: L. A. WILLOTT, Hunting Percival Aircraft Ltd., Luton Airport.

MANCHESTER
President: Sir ROY H. DOBSON, C.B.E., J.P., Hon.F.R.Ae.S.
Chairman: C. E. FIELDING, O.B.E., F.R.Ae.S.
Hon. Secretary: J. A. E. WATERFALL, 56 Manor Avenue, Ashton-on-Mersey, Cheshire.

MERTHYR TYDFIL
President: W. R. MORGAN.
Chairman: J. INGRAM.
Hon. Secretary: G. M. MAGILL, A.F.R.Ae.S., Teddington Controls Ltd., Cefn Coed, nr. Merthyr Tydfil, South Wales.

PRESTON
President: Sir GEORGE H. NELSON.
Chairman: W. HARPLEY.
Hon. Secretary: E. LOVELESS, A.F.R.Ae.S., Aircraft Division English Electric Co. Ltd., Warton Aerodrome, nr. Preston.

READING
President: Sir FREDERICK HANDLEY PAGE, C.B.E., [Hon.F.R.Ae.S.
Chairman: R. J. FENNER, A.F.R.Ae.S.
Hon. Secretary: G. H. DAVIES, Grad.R.Ae.S., Handley Page (Reading) Ltd., The Aerodrome, Woodley, Reading.

SOUTHAMPTON
Chairman: H. C. SMITH, F.R.Ae.S.
Hon. Secretary: S. F. STAPLETON, A.F.R.Ae.S., Folland Aircraft Ltd., Hamble, Hants.

WEYBRIDGE
President: G. R. EDWARDS, C.B.E., F.R.Ae.S.
Chairman: H. H. GARDNER, F.R.Ae.S.
Hon. Secretary: E. G. BARBER, A.R.Ae.S., Vickers-Armstrongs (Aircraft) Ltd., Weybridge.

YEOVIL
President: E. C. WHEELDON.
Chairmen: D. L. HOLLIS-WILLIAMS, F.R.Ae.S., and Dr. E. W. STILL, F.R.Ae.S.
Hon. Secretary: L. A. LANSDOWN, A.F.R.Ae.S., Westland Aircraft Ltd., Yeovil.

NOTICES

An annual sum of £250 is available for premium awards for papers, including Technical Notes, published in the Journal. Members and non-members of the Society are invited to submit papers on any aspect of aeronautics

Canadian Aeronautical Institute

The following Members of the Society have been elected Fellows of the Canadian Aeronautical Institute:—

I. M. Hamer (*Associate Fellow*), Air Vice-Marshal A. L. James (*Fellow*), B. S. Shenstone (*Fellow*).

Associate Fellowship Examinations, December 1956

Entry forms for Part II of the Old Syllabus of the Associate Fellowship Examination may be sent in up to 31st August 1956. This is the last occasion on which papers in the Old Syllabus will be set.

Entries for Part I and Part II of the Revised Syllabus closed on 30th June 1956.

News of Members

Dr. K. G. Bergin (*Fellow*) has been appointed Medical Superintendent, Air Services, of British Overseas Airways Corporation, and has also been elected a Vice-President of the Aero-Medical Association for the year 1956-1957.

W. E. Coe (*Graduate*) has taken up an appointment as a Performance Engineer with British European Airways at their Engineering Base, London Airport.

E. Fell (*Graduate*), formerly with Joseph Lucas (Gas Turbine) Ltd. Burnley is now employed as a Project Engineer with the Bristol Tabulatory Machine Co. at Letchworth.

A. J. Goodwin (*Graduate*) has been appointed a Development Engineer with A. V. Roe & Co. Ltd. at Manchester.

L. J. Green (*Associate Fellow*) is now employed as Assistant Chief Stressman at the Eton office of Condesco (Production & Industrial Designers).

E. E. Happé (*Associate*) has been appointed a Designer (Aircraft) with Avro Aircraft Ltd., Ontario.

J. A. Harrington (*Associate*) has left Field Aircraft Services, Bovingdon and has taken a Technical Liaison position with de Havilland Propellers Ltd., Hatfield.

M. A. Hollingsworth (*Graduate*) has been awarded a Ph.D. at the University of Southampton and will shortly be taking up an appointment in the Wind Tunnel Section at Vickers-Armstrongs (Aircraft) Ltd., Weybridge.

Dr. Theodore Von Kármán (*Honorary Fellow*) has been honoured with the decoration of Grand Officer of the Order of Orange-Nassau by the Dutch Government.

J. E. Lauderdale (*Associate Fellow*) has been appointed Design Engineer with the Lockheed Aircraft Corporation, Burbank, California.

F. W. H. Ponsford (*Student*) has been appointed Technician at Armstrong Siddeley Motors Ltd., Coventry.

M. L. Reed (*Associate Fellow*) has left Hawker Aircraft Limited and is now Senior Stressman at Aviation and Engineering Projects Ltd.

W. F. Shilling (*Associate Fellow*) has taken up an appointment as a Senior Design Engineer with the Atomic Energy Authority.

B. Towler (*Associate*) is now employed as an Airline Pilot with Pacific Western Airlines, Vancouver.

T. Wayne (*Associate Fellow*) formerly with Rolls-Royce, Ltd., Derby, has moved to Rolls-Royce of Canada in Montreal as Assistant Sales Manager (Aero).

M. J. Webber (*Graduate*) has left the Bristol Aeroplane Co. Ltd. and is now with Eagle Aviation Ltd. as Pilot (First Officer).

Institute of Physics—Joint Conference

This Conference is to be held at the College of Aeronautics, Cranfield, on 19th, 20th and 21st September 1956.

As the number which can be accommodated is strictly limited, applications should be made as soon as possible to: the Secretary, Institute of Physics, 47 Belgrave Square, London, S.W.1.

Subject to minor modifications, the programme will be as follows:

Wednesday 19*th September* 1956.

Day
- Opening Address—E. T. Jones, President of the Royal Aeronautical Society.
- "Experimental Methods in Kinetic Heating Tests"—W. H. Horton.
- "Temperature Distribution in Aircraft Structures, and the Influence of Physical and Material Properties"—K. L. C. Legg and G. F. Stevens.
- "Transient Thermal Conditions and Associated Stresses"—J. A. G. Holmes.
- Visit to College of Aeronautics Laboratories.
- "Extensometry"—R. J. Wilkins.
- "Pneumatic and Mechanical Extensometers"—S. S. Gill and J. H. Lamble.
- "Calibration and Modification of a Highly Sensitive Optical Extensometer"—F. J. Tranter.

Evening Film Show.

Thursday 20*th September* 1956

Day Symposium on Jet Efflux—introduced by:

M. O. W. Wolfe	D. James
Dr. R. Franklin	Dr. B. L. Clarkson
D. J. Mead	K. W. Hetzel

"Stress Analysis in the U.S.A."—A. F. C. Brown.

Evening Annual General Meeting of the Stress Analysis Group. Institute of Physics members only.

Friday 21*st September* 1956

Day
- "Crack Propagation—An Introduction"—Dr. A. A. Wells.
- "Crack Propagation from Design and Experimental Aspects"—C. R. Fletcher.
- "The Propagation of Fatigue Damage measured by Periodical Polishing"—B. K. Foster.
- "Strain Distribution in Notched and Cracked Aluminium Alloy Sheets"—V. M. Hickson.
- "Some Experiments on the Compensation for the Effects of Temperature on Strain Gauges"—R. D. Trumper.
- "Digital Methods of Bridge Balancing"—W. J. Walker.
- "Improvements Relating to Flight Recordings and General Instrumentation Techniques"—J. Arrowsmith.

The Conference will end at about 5.00 p.m.

News of Members—Continued

M. R. Williams (*Student*) has been awarded the S.B.A.C. Prize for studies in Aircraft Propulsion at the College of Aeronautics, Cranfield.

H. M. Woodhams (*Fellow*) is the first recipient of the Society of Licensed Aircraft Engineers' Silver Turnbuckle.

Elections

The following is a list of new members and transfers of membership of the Society:—

Associate Fellows

William Frederick Adams
Alexander Edward Anderson
Louis Sylvio Armandias (*from Companion*)
William Brian Axford
George Beardshall
Vishwa Nath Bhardwaj (*from Associate*)
Gerald Harrison Binns
Frederick Thomas Bolt
John Charles Francis Brunswick (*from Graduate*)
James Frederick Cook
John James Cripps (*from Graduate*)
William Davis (*from Associate*)
John J. Eden
Gordon Benjamin Evans
Edwin Robert Harold Goodfellow
Albert Harold Norman Herbert (*from Graduate*)
Arie Hillel (*from Graduate*)
Rodney Graham Hinkley (*from Graduate*)
John Charles Hodge (*from Graduate*)
Kenneth Holloway
Gerald Derek Horne (*from Student*)
David Hobson Hughes (*from Graduate*)
Edward Vernon Hullah (*from Graduate*)
Michael John Debenham Inskip (*from Associate*)
Gilroy Reginald Ketley (*from Graduate*)
Mangho Jeramdas Kirpalani
Leonard Leach
John Mansell Lewendon (*from Graduate*)
David Edward Lloyd
Harold Moore Lofthouse (*from Graduate*)
Denis Lawrence Lofts (*from Graduate*)
Bernard David Longbottom (*from Graduate*)
William Donald McCourty
Eric Jesse McKenna (*from Graduate*)
George Cameron Reid Mathieson
Charles William May (*from Graduate*)
Hugh Christopher Henry Mereweather (*from Graduate*)
Douglas William Fox Milestone
Derek George Mitchelson (*from Graduate*)
Henry Maxwell Nerriere
John William Paulsen (*from Graduate*)
Harry Rewitsch (*from Graduate*)
Stanley Joseph Sealey (*from Student*)
Wilfred Slade
Neil Sullivan
George Edwin Thirlwall (*from Graduate*)
Philip Rupert Thompson (*from Graduate*)
George Young (*from Graduate*)

Associates

Montagu William Norris Abbott
Ronald Allen
Arthur Laurence Barbrook (*from Student*)
Walter Edley Barker
Phillip Michael Bartley
Frederick George Beagle
Charles Henry William Champness
Ian Chichester-Miles (*from Student*)
Nigel Neville Ducker
Wallace Vaughan Galen
Grahame Kenneth Gates (*from Student*)
Richard Thomas Gibbons (*from Companion*)
Maurice Gough
Ernest James Groves
John Gunn
John Percy Halfey
Victor F. G. Hudson
John Anthony McColgan
Spencer Mills
Horace Odell
William Trelawney Lester Reed
Alexander William Robson
Hugh Guyer Small
Edward Frank Smith
William Kenneth Smith
Roy Fred Robert Storey (*from Student*)
Harry Townley
Derek John Whitehead

Graduates

David Ernest Hall Balmford
Roy Murray Braybrook (*from Student*)
Robin Michael Dakin (*from Student*)
David Ivor Davies
Derek Harry Freeston
Ronald Victor Holley
Paul Bertram Jarvis
Colin Reeves Jones
Sasson Yacov Katan
Ronald George King
Norman Geoffrey Marcus
Gordon Povah
Donald Smart Pringle
Denis Charles Roberts (*from Student*)
Michael James Somers Somervell
Brian Albert Stead

Students

Ian Malcolm Aitken
Roy Scott Elsworth Bristow
Charles Jonathan Britton
Robert Keith Burton
Ian Geoffrey Campey
Alan Alexander Clark
John Hubert Culhane
Robert Drummond Cutts
Kenneth John Davies
Rex Alec Kenneth de Winton
Derek Albert Edward Guttridge

Students—Continued

Christopher Barton Land
Martin Henry Lewin
William Richard Francis Metters
Alexander James Grant Michie
Michael Terrance Murphy
Ian Andrew Paul
Paul Francis Read
Adrian Gregory Reynolds
Kenneth John Riley
Narendra Sehgal
James Barrie Whitham
John Geoffrey Wright

Companions

David Dunbar Carrow (*from Graduate*)
Samuel Norwood Grant-Bailey

Graduates' and Students' Section—Visits

Boscombe Down

It has now been arranged for a party to visit Boscombe Down on Wednesday 29th August 1956. The party has been composed from applications of which there have been very many, in answer to the provisional notices in the June and July Journals. Unfortunately no further applications can be accepted due to limitation of numbers.

National Physical Laboratory

A visit has been arranged to the N.P.L. at Teddington for the morning of Saturday 22nd September 1956. Applications for this visit should be addressed to the Hon. Visits Secretary, N. K. Benson, 14 Wakering Road, Barking, Essex.

Aviation in Russia—1911

There is, in the Library, an interesting catalogue of the "First International Aircraft Exhibition in St. Petersburg 1911" This includes a series of articles on Aviation in Russia and if any member is sufficiently interested to translate them the Librarian would gladly lend him the catalogue to enable him to do so. Although the text is some 70 pages long, the print is large and frequent illustrations cut down the reading matter considerably.

Anglo-American Conference Proceedings—1955

The Proceedings of the Fifth Anglo-American Aeronautical Conference, which was held in Los Angeles in June 1955, are now available from the offices of the Society, 4 Hamilton Place, London, W.1, at £6 6s. 0d. per copy, including postage and packing.

The volume contains the 18 complete papers and discussions presented during the technical sessions of the Conference.

Changes of Address or Appointment

To assist in keeping the records of members correct and up to date the Secretary will be glad if all members will notify him as soon as possible of changes of address. He would also like to know of any change of appointment.

When notifying changes please give the following particulars:—

Name (in block letters). Grade of Membership.
New address (in block letters). Old address.

New appointment—Please give name and address of employer and position held.

Changes of address must be received *before the 15th of the month* in order to be effective for the Journal for the following month.

Journal Binding

Permanent Binding

The new prices for permanent binding of Journals are: 1955 Volume (including packing and postage) £1 0s. 0d. Previous Volumes (including packing and postage) .. .. £1 1s. 6d.

Journals, with a note of the name and address of the sender, should be sent direct to The Lewes Press, Friars Walk, Lewes, Sussex, and the remittance to the Secretary at the Offices of the Society.

Self-Binder Cases

Self-Binder cases of the "Easibind" type to hold 12 Journals (cost 11s. 6d. each) are available from the offices of the Society.

The Journal of the Royal Aeronautical Society

WITH WHICH IS INCORPORATED THE INSTITUTION OF AERONAUTICAL ENGINEERS

VOLUME 60 AUGUST 1956 NUMBER 548

The First Chadwick Memorial Lecture

The Life and Work of Roy Chadwick

by

H. ROGERSON, M.B.E., A.M.I.Mech.E., F.R.Ae.S.

The 987th Lecture to be given before the Society was the 24th Main Lecture to be given at a Branch of the Society and was also the First Chadwick Memorial Lecture. The Lecture, inaugurated by and held under the auspices of the Manchester Branch was given by Mr. H. Rogerson, M.B.E., A.M.I.Mech.E., F.R.Ae.S., before a large and distinguished audience on the 21st March 1956, at the College of Technology, Manchester.

MR. C. E. FIELDING O.B.E., F.R.Ae.S., Chairman of the Manchester Branch: It was his great pleasure to welcome to Manchester the President of the Society, Mr. Rowe, and Members of the Council of the main body. They were particularly pleased to have the President with them because, over the past number of years, he had taken a great interest in the Branches and they were very grateful for the help that he had given to them. They also had with them the President-elect, Mr. E. T. Jones, and he would like to say how very pleased they were to see him. They were also very glad to have with them the Chairman of the Branches Committee, Mr. Handel Davies who was the "father confessor" for the Branches now. In addition two other members of the Council, Sir William Farren, who was a Past-President and Sir Arnold Hall, were present and they extended a welcome to them.

The President and Council had elected to make this First Chadwick Memorial Lecture one of the Main Lectures of the Society and he therefore had very great pleasure in asking the President to take the Chair.

MR. N. E. ROWE, C.B.E., B.SC., F.C.G.I., M.I.Mech.E., HON. F.I.A.S., F.R.Ae.S., President of the Society: It was a great privilege to preside at the inaugural Chadwick Memorial Lecture. It was also a very great personal pleasure because —as so many of them became, he thought—he was a friend of Roy Chadwick who was a man of deep human qualities besides being a very great man in other ways. Chadwick had that spark of humour which made such a great difference and he was interested in people.

It was always a pleasure for him to come to the Branches. He had found it a great stimulus and a great personal pleasure to work with the Branches because they included some very enthusiastic people—they were young men; the Branches did a great work for the Society which he was sure would continue and grow as the years went on.

They inaugurated the Memorial Lectures at the Branches, he thought about four years ago, with the Rex Pierson Memorial Lecture at the Weybridge Branch and that was followed by others; the Mitchell Memorial Lecture at the Southampton Branch, the Barnwell Memorial Lecture at the Bristol Branch; the Sir George Cayley Lecture at the Brough Branch; this year two more of these Memorial Lectures were inaugurated; one at the Yeovil Branch, the Henson and Stringfellow and the other, the Pilcher Lecture, at the Glasgow Branch. Now they had the Chadwick Memorial Lecture. These were occasions to commemorate great men of the locality; they were also great men of the nation but particularly they were remembered as the people who did great things locally and therefore they were rightly celebrated at these Branch occasions. They hoped also that these evenings would stand out as highlights in the life of the Branch and that the Branch would always endeavour to make this particular lecture in the year's session *the* lecture of the year.

Their lecturer that evening was well known to them all. He again was an old friend. He knew Roy Chadwick from 1906. He was associated with him in the Avro Company for very many years as a friend first and as a fellow worker; he was sure they could have no one better to give this first Lecture.

I AM WELL AWARE that there are others still active who are better able to recount Roy Chadwick's early activities with Avro's, particularly I think of Sir Alliot Verdon Roe, Sir Roy Dobson and Harry Broadsmith—others of his early colleagues and mentors, John Lord, H. V. Roe and R. J. Parrott are no longer with us.

However, from an intimate friendship of 40 years and working for and with him—particularly I stress *with* him for 30 years—I will endeavour to outline what seemed to me the outstanding steps in his career, with Avro's and always faithful to Avro's.

It is inevitable that some reminiscences should arouse nostalgia, but I will try to make this paper as factual as possible, and I must crave indulgence if I refer to some eminent gentlemen by the names or initials by which they were known to us in those early days.

Roy Chadwick was born on 30th April, 1893, so that he was but a boy when the first pioneers were developing the art and science of flying, but as a young schoolboy he had one interest only and that was fixed-winged flying machines, and his heroes were the two Wrights, S. F. Cody and A. V. Roe. As boys he and I sat side by side in the local Urmston Church Choir, and

FIGURE 1. Roy Chadwick, aeroplane designer.

while most of us appeared to give rapt attention to the half hour sermon, Roy was usually sketching the " Argosies of the Skies " of his boyish dreams, often followed by a gentle reprimand from our choirmaster or the padre.

In the middle 1920's he tried to work up an enthusiasm for rotating wing aircraft, but failed to convince himself and concentrated therefore on fixed-wing " machines "; for complicated machines aeroplanes have become, making aeroplane design a " science " although " art " still contributes, and in my opinion and that of others better qualified, an aeroplane that is right, looks right, and the modern aeroplane is a thing of beauty to appeal to the aesthetic eye of the bird lover.

Before he devoted the whole of his working life and almost all his leisure hours to aeroplane design, Chadwick was quite a fair artist and violinist, and I have no doubt this was evidenced in a secondary way in many of his designs. After leaving school he took a post in the drawing office at the British Westinghouse, where his father was a departmental chief, and came under the very encouraging influence of his chief draughtsman G. E. Bailey (now Sir George Bailey), and later in the machine shops with Tom Fraser, who during the 1939-45 War controlled the manufacture of hundreds of Lancasters at Metropolitan-Vickers.

During the period before the First World War Chadwick eagerly followed the fortunes of the local pioneer, A. V. Roe, and eventually persuaded A. V. to take him on as a draughtsman under R. J. Parrott at Brownfield Mills, Newton Heath, Manchester, in 1911. His unbounded enthusiasm and unsparing energy soon won A. V.'s approval and A. V. " fathered " him, rather than employed him, in those early days before the 1914-18 War, when orders were few and money scarce. He also found a good friend in Reg. J. Parrott, then Chief Draughtsman to A. V.

At the time when Roy Chadwick entered the design office the general lines of the Avro tractor biplane were in process of being stabilised, although A. V. still had a great regard for the monoplane lay-out as evidenced by his enclosed cabin monoplane design in 1912, but I look upon the first few years as covering the development of the Avro 500 which was built in 1912. On the surface it may be thought that there is little scope for originality in the development of a type, but those who have been intimately concerned with design will no doubt agree that many of the most successful aeroplane designs have been the direct outcome of development of perhaps less successful prototypes.

The result of this was that Chadwick's first few years were " formative " ones, during which he began to appreciate the necessity for pleasing the user—at this period the flying instructor—and he would spend a large amount of design time on the cockpit lay-out and the harmonising of the three flying controls.

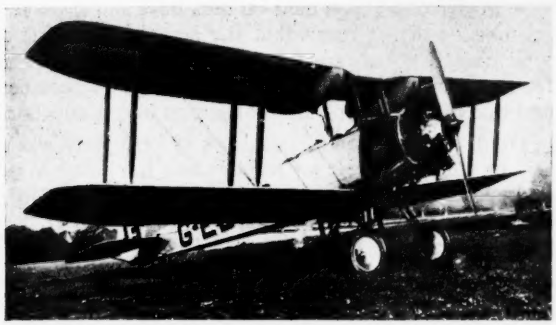
Considering the Avro 500 of 1912 as the prototype of the 504 series, this development period continued almost unbroken until 1924, when the 504N with the Armstrong Siddeley Lynx Engine was put into production for the R.A.F.—an amazing life of a basic design which was an accepted trainer from 1914 until about 1930.

In the early days of the 1914-18 War the 504K performed many roles other than Flying Training—particularly I call to mind the bombing of the Friedrichshafen Airship sheds on 21st November 1914 and successful Zeppelin straffing in early 1915—but in 1916-17 the 504J had made such a name for itself as a trainer that, when the Gosport system of training was evolved by the late Col. Smith Barry in 1917, the 504K was standardised as the trainer for the Allied Air Forces, and the demand was so great that in 1918 the programme was for 500 aircraft a month, of which the parent firm had to produce almost half, the rest being produced by other Air Ministry sub-contractors.

FIGURE 2. " Flying trainer " Type 504K.

FIGURE 4 (*right*). Twin-engined bombers Type 523.

FIGURE 3 (*below*). "Flying trainer" Type 504N. (Oleo undercarriage)



While the 504 design was being finalised in 1916, A. V. made schemes for a twin-engined bomber with the front gunner and bomb aimer in the nose in front of the pilot, and a gunner with a rotatable gun mounting just aft of the main planes, again of biplane lay-out. The detail design was left to Chadwick while A. V. concentrated on the difficult task of evolving a horizontal tier-stowage of the bombs.

In 1916 the firm built an Experimental Shop and Aerodrome at Hamble, the quaint yachting rendezvous on Southampton Water, and to which the Design Office was transferred.

FIGURE 5. From left to right, Raynham, Thurston and Chadwick with Type 529.

This design passed through many phases with different engines and armament. It will be noted that on the first three shown in Fig. 4, the power eggs were fitted approximately midway between the upper and lower wing clear of the wing surfaces, although Chadwick was not himself convinced that this was the optimum position, and on later developments built in 1917 (Type 529A), two 210 b.h.p. engines were fitted on cradles on the upper surface of the lower planes.

Still further developments in 1918 were fitted with 300 h.p. Siddeley Pumas and ABC Dragonfly Engines, again fitted on cradles on the lower plane. It is worthwhile mentioning here that the optimum positioning of power units received a great deal of Chadwick's attention as evinced in his most outstanding success, the Lancaster, where he disregarded advice regarding the best vertical position of the engines for minimum drag at top speed, and adopted a position more favourable at a higher lift coefficient, which turned out to be good policy as the loaded weight and operational altitude went up.

A two-seater fighter, Type 530, was produced in 1917 incorporating drooping trailing edges, a forerunner of the landing flaps of two decades later, but it must be admitted that, in the opinion of many, this aircraft was not as good as Barnwell's famous "Brisfit" and it did not go into production.

As an indication of his versatility, Chadwick had considered the importance of rapid manoeuvrability for the single-seater fighter which had become standardised towards the end of the 1914 War, and he did not

FIGURE 6. Twin Dragonfly Manchester Type 533.

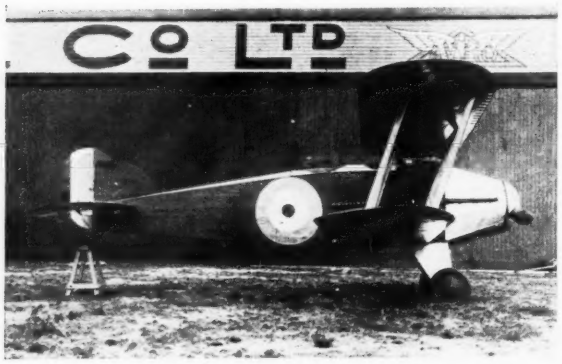


FIGURE 7. Twin-seater fighter Type 530.

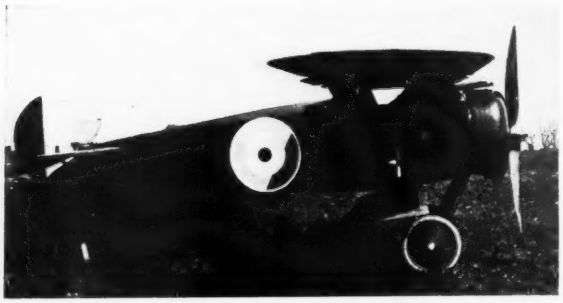


FIGURE 8. Rigid braced " Spider " fighter Type 531.

FIGURE 9. Type 534 " Baby " with its designer.

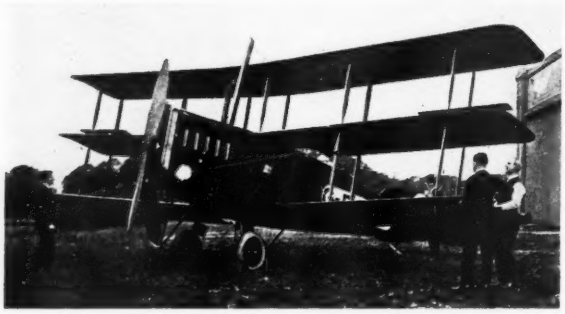


FIGURE 10. Puma passenger triplane Type 547.

subscribe to the generally adopted theory that this must be obtained by reducing the distance between the c.g. and the neutral point, *i.e.* bringing the c.g. back. Rather did he favour a fairly stable aeroplane with very rigid structure coupled with powerful and well harmonised controls. To test his ideas he designed a little $1\frac{1}{2}$ plane fighter with welded steel Warren girder interplane bracing, which was built as an unsponsored Private Venture and fitted with a " borrowed " 110 h.p. Le Rhone engine. This turned out to be a delightful aeroplane to handle, extremely manoeuvrable and without " tricks " or " vices," but our methods of trying to obtain official approval were rather unorthodox and were frowned upon. Various well known fighter pilots were invited to fly it on their off duty days and asked for opinions, with the result that the aeroplane became a topic of general conversation in the messes and came to the notice of the heads at the Air Ministry. I need not add that this aeroplane also did not go into production.

With the cessation of hostilities in 1918 there was a serious glut of aircraft for training and few pilots to be trained, in fact there were some thousands of 504K's thrown on the market, including of course, many that had not been made by Avro's.

Hopes were high of receiving orders from Canada and other overseas countries, but firm orders were not forthcoming although many specifications were accepted and tendered for. It appeared that other countries besides ourselves found themselves short of cash, but still wished to keep up to date with our design trends. The result of this was that, while our business chiefs maintained a small but steady income from joyriding, A. V. and Chadwick longed for the re-introduction of what, for want of a better name, I call low-powered flying.

A. V. favoured a return to wing loadings of the order of 3 lb. per sq. ft. and powers of about 20-30 h.p. on aeroplanes of about 600 lb. all-up weight, which could be flown at 40 miles an hour within the boundaries of a large field, if desired purely for the sheer joy of flying. Chadwick wished to go what he considered one stage farther, and design an aeroplane suitable for cross-country touring with rather heavier wing loading and about 40-50 h.p.

Now, 37 years later, I am not quite sure which was right, in fact there still seems to be an outlet for both types.

As in many cases the availability of power plant decides the issue, there being no very light motorcar or cycle engines suitable for the ultra light aeroplane, whereas A. V.'s 35 h.p. Green engine of 1910 had been preserved intact by Mr. Fred May of the Green Engine Co. and the whole of the manufacturing drawings were still available.

The outcome was that the Avro " Baby " was designed and made round this overhauled engine, and a delightful little aeroplane it turned out to be, cruising at about 65 m.p.h. with a fuel consumption of about one gallon for 35 miles. Such was its success that a small production line was laid down at Hamble, and Peter Brotherhood's of Peterborough undertook the

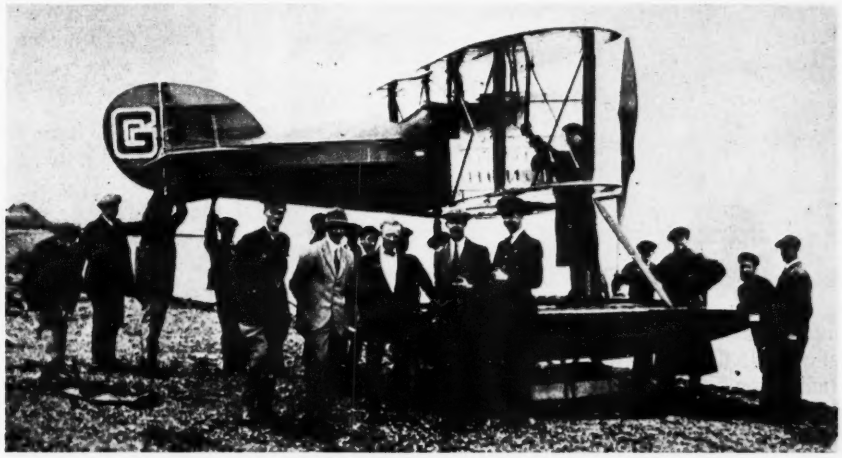


FIGURE 11. Schneider Trophy racer Type 539.

manufacture of the engines. The co-operation between the airframe and engine firms was greatly assisted by Mr. Fred May spending his week-ends on his luxurious motor cruiser, which was anchored in the Hamble river near the Avro works.

The " Baby " was a staggered biplane of 25 ft. span and a gross weight of just under 900 lb., and as proof that it was a practical touring vehicle, we may instance the winning of the Aerial Derby Handicap Race in 1919 and 1920, and Bert Hinkler's non-stop flights from Croydon to Turin and Riga, and later from Sydney, Australia, to his old home town Bundaburg, where he landed in the main street. This last flight was of 800 miles non-stop, which was good going 36 years ago. Shortly after, Hinkler also piloted the Baby to success in the Grosvenor Challenge Cup Race.

It might not be out of place to put on record the not unimportant part played by Hamble from 1916 onwards—I call to mind Faireys testing a seaplane with a Rolls engine with a two-speed gear ingeniously contrived from the epicyclic reduction gear, and at the end of 1917 a hush-hush aeroplane from Farnborough that we used to call, perhaps disrespectfully, " Bill Farren's Futuristic Flying boat." (About this aircraft we had long arguments as to whether it was a flying boat or a single float seaplane, as the tail unit was on outriggers and not on the hull). Also of course many Schneider Trophy Racers were put through their paces over Southampton Water.

The " Baby " was also fitted with floats but, because of the low power, was difficult to unstick from smooth water and a two-seater land version was built, which Hinkler and I took to 11,000 ft. without experiencing " wallowing."

After the Armistice Chadwick had become an efficient pilot under Allan Hammersley's expert tuition and, whenever an opportunity occurred, used to cruise round the country in the works " Baby " until on one such flight he crashed into the garden of Hamble vicarage, then occupied by A. V.'s brother the Rev. Everard Verdon Roe. Although the crash was from a low altitude, Chadwick sustained multiple injuries which very nearly cost him his life. I well remember visiting him in a nursing home in London, and knowing the extent of his main injuries enquired after each one individually. The following is typical of the conversation which took place.

" Have they fixed your arm up? "

" Yes, the broken ulna and the radius have been repaired by silver plates and rivets."

" And the smashed top of your femur? "

" Oh they have fastened that together with four-inch woodscrews."

" And what about the kneecap which was in four distinct pieces when I visited you in the Southampton Hospital? "

" Oh that has been fixed up with wire " rivets " just like they repair household pottery."

However, such was his vitality and urge to live that he returned to his task with fresh energy, but promising to give up solo flying.

In an attempt to revive the triplane configuration, 504K wings and other components were incorporated in a passenger-carrying triplane, the 547, in 1919. Two prototypes were built and the second, fitted with a Puma engine, was entered in the Passenger Aeroplane Competition then taking place. It was flown by Allan Hammersley, but was not successful, due mainly to its being neutrally stable whereas one of the tests was to prove ability to fly " hands off " for a reasonable length of time.

In fairness to Chadwick it must be pointed out that he was out of action in hospital during the development of this type.

The revival of the Schneider Trophy Race in 1919 gave Chadwick the opportunity he had long wished for to design what at the time was a really fast single-seater. (Fig. 11). This design was powered by a 300 h.p. Armstrong Siddeley Puma six-cylinder in-line engine, but was chosen as a reserve aeroplane to the faster and more powerful Sopwith seaplane.

This race was held at Bournemouth and was marred by very foggy weather and other misfortunes. The Sopwith had damaged its floats in practice and could not be repaired in time for the race, but in order to add

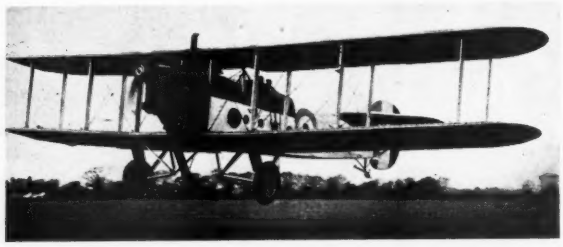


FIGURE 12. " Aldershot " bomber Type 549.

interest to the race, Hammersley, on the Avro, was allowed to fly unofficially as he had not been nominated. Janello on the Savoia was first past the post but had been unable to see all the turning points, and was disqualified for lack of proof that he had rounded them all.

The Type 539 was later revived as a landplane with a Napier Lion engine with the Aerial Derby in view, but unfortunately it crashed irreparably on its test flights due to no fault of the aeroplane. This was a great disappointment as we were all sure it would have given Folland's Gloster " Bamel " a very good run for its money.

In 1920 the Air Force was being re-equipped to some small extent, and the single-engined Avro " Aldershot " Bomber was Chadwick's first contribution. The prototype was largely of steel tubular construction and was fitted with the new Rolls " Condor " of 650 h.p. with their well tried epicyclic reduction gear, but the production models had the later Condor III with spur reduction drive to the propeller. An ambulance version styled the " Andover " was also produced, and one aircraft was later used as a flying test bed for the 1,000 h.p. Napier Cub and for the ill-fated " Typhoon " engines. The Cub version, flown by Bert Hinkler on its initial flight, was at that time the most powerful single-engined aeroplane in the world.

The reputation gained by the Aldershot was maintained by the " Bison," a Lion-engined Gunnery spotter for the Fleet Air Arm. To give the pilot optimum vision when landing on the deck he was positioned high up in front of the top plane, and the " forrard deck " sloped sharply down to the nose.

FIGURE 14. " Ava " bomber and torpedo carrier Type 557.

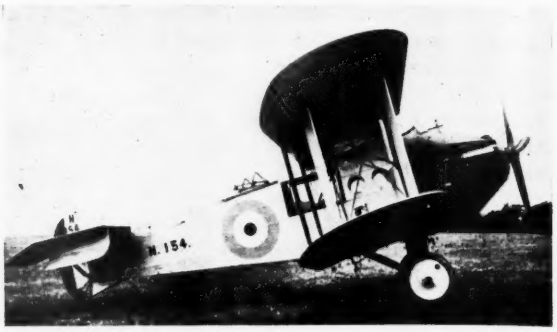


FIGURE 13. " Bison " fleet gunnery spotter Type 555.

Pilots had for some time been accustomed to having a horizontal section of fuselage nose in front of the cockpit to " con " by on approach and landing, and after many discussions we satisfied Hinkler by fitting a 4 ft. wooden rod horizontally on outriggers in front of the windscreen for the initial flights. Happily, Hinkler asked us to remove it after the first two flights, and the Navy Pilots soon got used to the angle of the foredeck and appreciated the excellent view provided when approaching an aircraft carrier, although I personally saw few three point landings—which we always hoped for as aircraft were not then fitted with brakes.

Chadwick's next design was for a large twin-engined bomber and torpedo carrier named the Ava, with Condor engines, which when first flown was, to my knowledge, the largest all-metal aircraft—at least in this country. It had a biplane tail and twin rudders, an all-up weight of about 10 tons and carried the large 21 in. torpedo with 500 lb. war head, but the introduction of an 18 in. torpedo with a 500 lb. war head made this size of aircraft unnecessary for torpedo dropping.

In 1923 an intensive effort was made to cater for the private owner who would buy an aeroplane for the

FIGURES 15 and 16. Motor cycle-engined ultra light aeroplanes Types 559 and 560.

FIGURE 17. " Avis " two seater light plane Type 562.

FIGURE 18. " Avian " club light trainer Type 594.

simple pleasure of flying, and for the first Light Aeroplane Competitions at Lympne. A. V. designed a monoplane and Chadwick a biplane. Air-cooled motor cycle engines developing 20 to 30 h.p. were fitted, and it is on record that Hinkler on the Blackburn-engined Monoplane won the prize for the longest distance flown during the competition, and Hammersley, on the Biplane fitted with a geared Douglas T.T. engine was joint winner of the Altitude prize.

It is interesting to note that to-day, after more than 30 years, the design of a light aeroplane for the private owner, who loves flying for its own sake, is still not settled and many consider that it is substantially discouraged by the complexity of the regulations which it has been deemed necessary to introduce over the intervening years.

A year later, in 1924, the design of the light aeroplane was carried a stage farther in the " Avis," powered by the twin-cylinder Bristol Cherub engine with a geared down propeller, but its success was marred by teething troubles with the prop-shaft gearing.

Continuing the development of the private owner's aeroplane the Avian was produced two years later. It was somewhat bigger and heavier and was a dual control two-seater fitted with one of Halford's Cirrus engines and it proved to be ideal for the Light Aeroplane Flying Clubs which the Government now decided to sponsor. The Avian appeared shortly after the early D.H. Moth, and many were the intense, but friendly, duels fought out at Race Meetings between Hubert Broad on the Moth and Bert Hinkler on the Avian with honours, I think, fairly even.

Among the outstanding flights made on Avians were:—Hinkler's solo flight to Australia in 15½ days, a record which stood for nearly 3 years until Kingsford Smith later covered the same route in 10 days on a more powerful and faster Avian. Nor must we forget Miss Winifred Brown's superb piloting of her Avian in the King's Cup Race of 1930, when she won the cup in the face of intense male competition.

An experimental braced Monoplane Avian was also built, but was not much faster than the standard biplane and did not go into production.

Meanwhile the basic 504 design was being further developed and, as the 504N with Lynx engine, still retained its position as the standard flying trainer of the R.A.F. The power had by now increased from 100 to 200 b.h.p., and the weight by 16 per cent., which many thought retrograde, but an attempt to revert to a lighter version with the revived 100 Mono engine met with only partial success in spite of its being named, very appropriately, the " Gosport " trainer.

Not everyone remembered " Gosport " and the rotary engine was looked upon as " messy " and was not welcomed by the new generation.

In 1925 an Air Ministry Order enabled Chadwick to design the " Buffalo," our second torpedo-carrying aeroplane, with the same type of construction as the Bison; wooden biplane wings and braced steel tube fuselage. It was sufficiently successful to earn the limited production order that the Treasury grants then allowed.

About this period development was proceeding in many firms on Monocoque Construction, and as a first

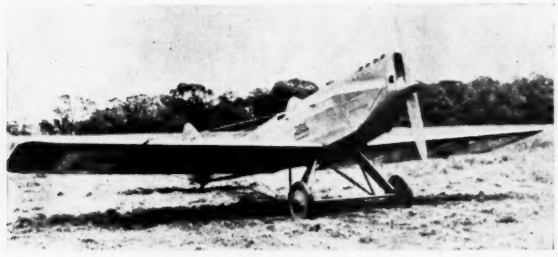


FIGURE 19. " Avian " monoplane Type 625.

FIGURE 20. " Buffalo " torpedo carrier Type 571.

FIGURE 21. " Avenger " experimental single seater Type 566.

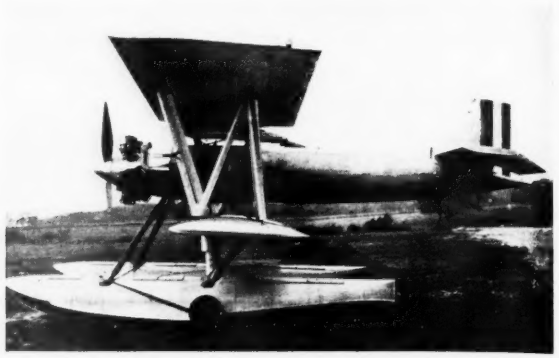


FIGURE 22. " Avocet " fleet fighter Type 584.

FIGURE 23. " Antelope " day bomber Type 604.

FIGURE 24. " Avro Ten " air liner Type 618.

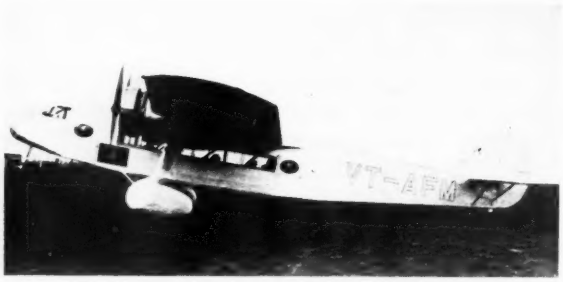


FIGURE 25. Four-engined monoplane Type 642.

FIGURE 26. " Avator " Imperial Airways charter aeroplane Type 652.

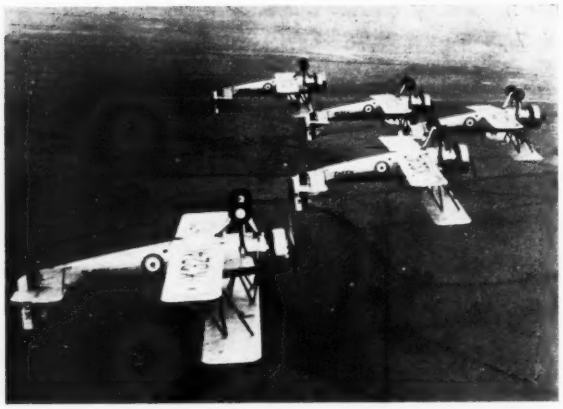


FIGURE 27. Tutors in inverted formation. Type 621.

FIGURE 28. Prefect advanced trainer Type 626.

stage Chadwick produced a design for a Lion-engined single-seater fighter with a wooden monocoque fuselage.

We considered the Avenger a beautiful aeroplane, but the design was really in the nature of an exercise and one of the valuable lessons we learned was the difficulty of providing accessibility, when compared with the braced strut fuselage.

The experience gained with the Avenger was incorporated in the "Avocet," two of which were built for the Ministry in 1926. The Avocet was an all-metal, single-seater fleet fighter powered by a supercharged Armstrong Siddeley Lynx engine, and had a Duralumin monocoque fuselage and Dural spars for the wings, which were quickly detachable for stowing below decks.

About this period it was the Ministry's policy to order a few prototypes of many designs, rather than standardise on a few types, in order to encourage initiative among the design teams; therefore the number produced to any one design must not be taken as a measure of its success.

Following this policy, the "Antelope" single-engined day bomber was produced in 1927 and again was mainly of Duralumin construction, but with a fuselage of rectangular section with Dural frames covered with Dural sheet.

It was during the design of this aircraft that Roy Chadwick and his colleague in Manchester, Roy Dobson, had many discussions about the advantages of Antony Fokker's welded tube construction, and an alternative design fuselage of welded tube construction was made for the Antelope. From a comparison of the two we learned that the welded tubular construction was singularly free from the modern bugbear — stress concentrations — and when J. D. Siddeley (the late Lord Kenilworth) took a financial interest in Avros in 1928, the first thing the two Roys did was to press for an agreement with Fokkers to obtain their "Know How" on welded construction.

This coincided with the Design Office moving from Hamble to the production works at Manchester, and there is no doubt that the close collaboration between the two Roys was vital to the success of the Avro welded construction, which is easy for the draughtsman but doomed to failure without the closest collaboration with the production staff. This was abundantly evident in later designs, such as the Avro Ten, which was virtually a redesigned and anglicised version of the successful Fokker F 7/3 M with three Lynx engines, and the smaller Avro Five Special Charter aeroplane with three Genet Major engines.

These monoplanes had thick wings of two-spar wooden construction with plywood covering, and were noted for their robustness as well as excellent flying qualities. The Ten carried 8 passengers and the Five carried four passengers, the number of crew bringing the complement up to the designation.

A successful development of the Avro Ten was the Avro 642 of 1933; it was somewhat larger, carrying up to 16 passengers and a crew of two, and was powered by two of the geared Jaguar engines which had been developed for Imperial Airways. A second model was fitted with four Lynx engines to the special orders of Lord Willingdon, then Viceroy of India, who used it extensively with an Avro Ten as a tender for his entourage.

This type of construction reached its peak in 1933 in the design of the 652, a low wing monoplane for Imperial Airways Charter work, and which achieved undying fame as the Anson of the Second World War, of which more anon.

In 1929 the Air Ministry decided at long last that the ubiquitous 504 should be replaced as the standard trainer after 12 years' superb service, and they instituted a competition which lasted for three years and in which many designs were thoroughly tested under service conditions. The outcome was the acceptance of the Avro Tutor as the standard type for *ab-initio* flying training.

On this design the fuselage was again of welded tubes, but the biplane wings had spars of thin steel strip recently developed at Armstrong Whitworth Aircraft Ltd., now an associate company. A development contract for Tutors called for the five cylinder Mongoose engine, but the production aircraft were fitted with the more powerful Lynx engine.

During this last decade the large increase of speed, combined with an inappreciable increase in wing loading, had caused many designers concern about the penalties to be paid for the large shift of c.p. at top speed and in the dive, of the cambered aerofoils then in general use and on the Tutor this was met by using an aerofoil with a reflexed trailing edge, namely R.A.F. 28; but, like many others, we soon realised that the ailerons of such aerofoils tended to float down as the speed increased and as a consequence the c.p. would wander back locally, offsetting the advantages of the reflexed section. However, on the Tutor, Chadwick had spent more time on the controls and cockpit lay-out than on the remainder of the design, and on this design control column circuit moved the ailerons up with the downward movement limited by the relatively stiff balance circuit.

The Tutor was outstanding in its manoeuvrability, and was often used in stunting demonstrations which were then popular at flying meetings on the Continent, where it competed against aeroplanes specially designed for the purpose.

A development of the Tutor was the 626 Advanced Trainer, designed mainly for export to countries who wanted their basic trainer to cover also bomb aiming, gunnery and photographic reconnaissance training. The R.A.F. used a number of these aircraft under the name "Prefect."

A further variation was supplied to the Egyptian Government to be used, I believe, in the war against drug smuggling.

The success of the Tutor as a trainer led naturally to schemes for a scaled-down version for the private owner and the Flying Clubs, and design work was precipitated by Sir Roy Dobson, in his secondary capacity as a super salesman, winning an order from

FIGURE 29. Club Cadet Type 638.

the Irish Free State on Chadwick's brochure, before the Design Office had really got going.

The outcome was the Cadet, powered by a Genet Major engine, which took the air about three months from this and, despite its being overshadowed by the Tutor, the demand from private owners and flying clubs, both here and abroad, kept it in production for six years.

Passenger-carrying cabin versions of both the Tutor and the Cadet were available and sold in small quantities.

From 1929 to 1934 it seemed that nearly every country was demanding a new design of trainer, and Chadwick's last design of Biplane Trainer was the 636 powered by a Panther engine and with a top speed of 230 m.p.h.

FIGURES 30 and 31. Commodore and Cabin Cadet Types 641 and 639.

The period from 1935 onward, covering the R.A.F. expansion period and the Second World War, was one of tireless activity in the Avro design team which can be but inadequately covered by a small section of this paper, but in many ways it represented the summit of Roy Chadwick's career and stamped him as one of the greatest aircraft designers in the world.

In 1933 Avros were invited to modify the Imperial Airways 652 design to make it suitable for Coastal Reconnaissance, and while this involved some very interesting and ingenious modification work which resulted in one of the most successful and well liked aeroplanes of all times, it was to Roy Chadwick still a development of an existing design and he always felt he could have done better if allowed time to design from scratch for one specific purpose.

However, Chadwick's flair for evolving aeroplanes adaptable for many roles was again in evidence, and the Type 652A "Anson" became the "Faithful Annie" first of Coastal Command and later, of the many sections of the Empire Air Training Scheme. Over 11,000 Ansons were built in this country and Canada, and one might say that it played a similar glorious, if unobtrusive, part in the Second World conflict which the 504 had played during and after the 1914-18 War.

A number of amusing incidents connected with Ansons come to mind which lightened the tedium of our deadly serious work. On one occasion a patrolling Anson was attacked over the sea by an Enemy fighter at about 5,000 ft. but the Anson pilot, confident of its remarkable manoeuvrability, dived to within 50 ft. of the sea before pulling out, while his pursuer, over-confident of an early kill, found too late that he could not pull out and continued on into the "drink."

On another occasion a pilot, on being attacked, put his Anson into a continuous tight turn which was about 50 per cent of the possible radius of turn of the attacking fighter. The pilot's account continued that the fighter tried in vain to get the Anson in his fixed gunsights while the two machines flew almost side by side in ever decreasing circles, until the enemy pilot grew tired of this "Ring of Roses" and decided to quit before running out of petrol.

FIGURE 32. High speed trainer Type 636.

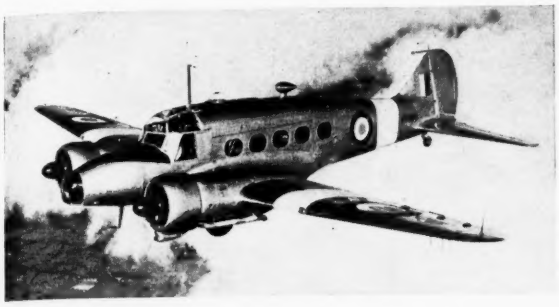


FIGURE 33. " Faithful Annie " Type 652A.

I should like to put on record a brief history of " Faithful Annie ": —

May 1934. Tender submitted and accepted in September.

March 1935. Prototype Anson flew.

March 1936. Ansons entered squadron service as Reconnaissance aircraft.

December 1939. Anson adopted for the Commonwealth Training Plan.

1940-1945. Nearly 10 million miles flown by the Anson Taxi service of the Air Transport Auxiliary.

27th May, 1952. The 11,020th and last, Anson delivered.

In 1936, while the Anson was being developed, Chadwick was devoting the greater part of his working hours to the design of what was to become a major contribution to Bomber Command's terrific offensive—a medium heavy bomber of long range.

Whereas on the Anson a wing loading of 20 lb. per sq. ft. was at first considered as high as we dared go for safe emergency landing, we were now forced to consider wing loadings at take-off of 40 to 50 lb. per sq. ft., and I well remember our being told that if a pilot were forced to land shortly after take-off, he was to keep his

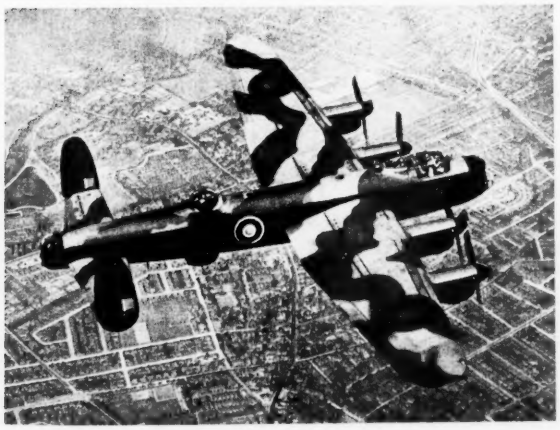


FIGURE 35. " Lancaster " bomber Type 683.

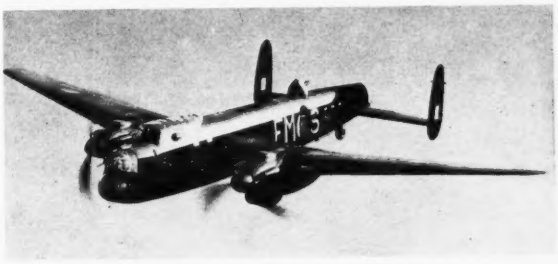


FIGURE 34. " Manchester " bomber Type 679.

undercarriage retracted, prepare for a belly landing and pray very hard.

In the light of later experience this may sound a very pessimistic outlook, but in 1936 that experience was lacking and it must be remembered that we were commanded to design aeroplanes which would bring the crew back alive, even if the aeroplane were badly damaged or the numbers built had to be reduced.

The design put forward was a twin-engined mid-wing monoplane powered by two of the new Rolls-Royce " Vulture " engines, and although any designer would prefer to fit well tried and developed power units to his prototypes, this is not always possible, and in this instance the demands of the technicians were too great to allow of the development of both aircraft and engine. However, this bomber, the " Manchester," made many bombing raids and, in Conversion Units, provided the very necessary experience in the operation of this new type of high wing loaded bomber, but it was completely overshadowed by its development, the " Lancaster " fitted with four of the well-tried Merlin engines. This was virtually a private venture put forward by the two Roys, who spent much of their time together on the shop floor discussing details with the men, resulting in a design both efficient in operation and economical to produce.

The outstanding feature of this design was undoubtedly the bomb-bay capable of accommodating almost any combination of bombs and mines. The

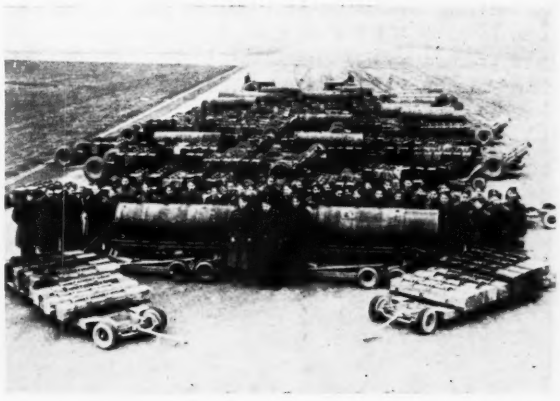


FIGURE 36. Bomb load for 20 " Lancasters."

FIGURE 37. " Lincoln " bomber Type 694.

FIGURE 39. " Tudor " air liner Type 688.

story of the Lancaster is, I hope, still too fresh in our memories to need repetition here and the film " The Dam Busters " will have shown the layman the type of operation it was ideally suited for. The flight testing of these aircraft at very low altitude provided Sam Brown with much " dicey " flying before the aeroplanes were handed over.

I think it is accepted that the Lancaster marked the peak of Roy Chadwick's career and stamped him as outstanding among the world's designers. If figures are required, 7,366 were delivered; they dropped 600,000 tons of bombs on the enemy and nearly 7,000 tons of food to our allies, apart from being used to repatriate 74,000 ex-prisoners of war, which is a record of service which will surely stand for all time.

When the Lancaster first went into operation a cruising altitude of 15,000 ft. was sufficient to defeat the Ack-Ack defences, but it soon became evident that this must be increased, and in the later stages of the war it was usual to cross the channel at 22,000 ft. A logical development was the " Lincoln," with increased span and with later mark " Merlins " to give a normal cruising altitude of 25,000 ft.

With the end of the Second World War in sight, Chadwick turned his attention to the development of the Lancaster and Lincoln for more peaceful uses. From the Lancaster came the " York." The York had the Lancaster wing, power units and undercarriage set high on a rectangular body, which gave maximum capacity for carrying paratroops, V.I.P.'s or freight. Yorks were used to carry V.I.P.'s on many historic occasions and gave yeoman service during the Berlin Air Lift; their ease of loading and unloading allowed more flights per day than any other aircraft. I have myself timed the loading of 15,000 lb. of cases of food in 20 minutes.

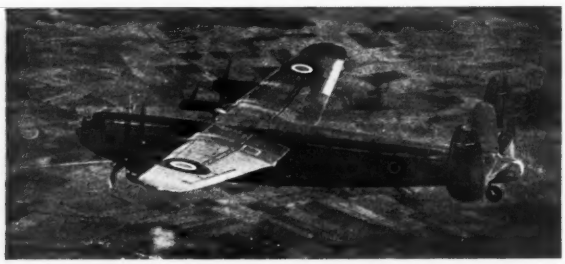


FIGURE 38. York transport Type 685.

At the close of the War a number of Lancaster fuselages were modified to make them suitable for long distance passenger service and, as the " Lancastrian," these aircraft inaugurated the England-New Zealand express service, Australia being reached three and a half days after leaving England.

Towards the end of the War the firm were asked to look into the design of a passenger-carrying aircraft for the Trans-Atlantic service, using as many Lincoln components as possible, with the object of staking a claim to the " prestige " trans-ocean routes as soon as possible after the end of hostilities.

Chadwick's reply to this challenge was the " Tudor," the first Mark of which flew a month after the war ended. I believe it was the first passenger aircraft to be built in this country with a pressurised cabin, which brought many new problems both with the structure and the complicated equipment.

The new word " comfortisation " was in everybody's mind, and what with the complications of pressurising a large cabin to give 8,000 ft. conditions at 25,000 ft. and heating and conditioning the large amount of air circulated, we often longed for the ideal system where passengers were given a safe dose of chloroform in a " Passing Out " chamber and then stacked on shelves in the Cabin, later to be " Passed In " on reaching their destination and handed over to relatives in a more or less normal condition.

The Tudor was used on the South Atlantic route to South America and its development continued through many marks, not without vicissitudes, well reported at the time. The Mark 8 was powered by four Nene jet engines, and a development of this, the Ashton, has done and is still doing remarkable high altitude research work.

About this time Chadwick's imagination was fired by news of the wind tunnel work recently done in Germany on Delta shaped aerofoils, and while others thought in terms of the small Delta fighters, he visualised a large 100 ft. span Delta wing thick enough to house all the turbo-jet power units and the fuel tanks, and with merely a slight swelling out at the centre for the crew's cockpit and the bombs.

A tender covering the scheme was submitted to the Ministry with the results we now know. Roy Chadwick did not live to know that Avros were being ordered to proceed with the design of the Vulcan, and as modern aeroplanes are not designed by one man but by a team, the greater credit for the Vulcan must be given to those who followed him.

He was killed when a Tudor II crashed just after take-off on a test flight on Saturday 23rd August, 1947, probably as he would have wished to finish, at the height of his fame, but a very sad end to one who surely had earned his share of relaxation as a reward for his endeavours and achievements. I hope it will be some consolation to Mrs. Chadwick and their two daughters to know that his memory is being perpetuated in this series of lectures.

Recognition of his achievements had come in many ways—His late Majesty King George VI conferred the C.B.E. on him, and Manchester University made him an Honorary Master of Science and an Associate of the College of Technology. He had long been a Fellow of the Royal Aeronautical Society and he was also a Fellow of the Royal Society of Arts.

As no successful aeroplane design can be the fruit of one man's effort but is the outcome of perfect team work, it might not be inappropriate to mention here some of Roy Chadwick's team in the early days at Hamble and of whom I have very pleasant recollections.

There was Frank Vernon, now a Professor of Engineering in Canada, Jock Ratcliffe now at Handley Page's, George Dowty, Rowland Bound and Charles Blazdell of Cheltenham, Clifford Tinson of Bristol's, C. H. Horrocks and Hugh Norwood now outside aircraft, and lastly, Jim Turner, George Martin and Ernest Brett still faithful to the Avro Design Office.

I have not included Sir Roy Dobson, although he was there, as he always insisted that his metier was producing and not designing aircraft.

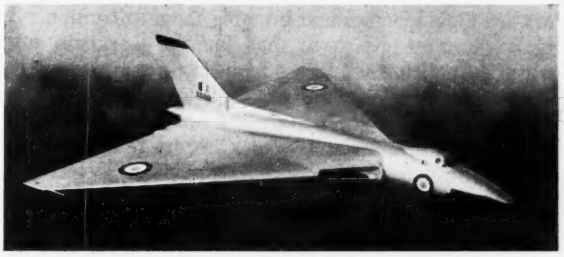


FIGURE 40. Avro " Vulcan " bomber.

I have tried to resist my natural inclination to be reminiscent and have endeavoured to place on record the facts, together with a little of the philosophy, behind the various designs as a tribute to a great designer whose prolific contribution to the development of aviation is second to none.

I looked upon Roy Chadwick as the complete designer in that he personally supervised every detail, spending the greater portion of his time in the Technical and Research departments, on the drawing boards and on the shop floor.

Whether this type of designer will disappear in this " transonic " and " supersonic " age I would not care to prophesy but if this is inevitable it is to be regretted as the passing of an era when aeroplanes were designed with personalities by personalities.

I should like to thank Sir Roy Dobson, Sir William Farren, Mr. C. E. Fielding and other colleagues at Avro's for suggestions and help in the preparation of this paper and particularly to Mr. Fielding for the tedious work of proof reading.

N. E. ROWE: We have heard a great paper splendidly delivered. There clearly is no general discussion on a paper of this sort but there are two present who can add, I think, a little to what has been said—Sir Roy Dobson and Sir William Farren.

SIR ROY DOBSON, C.B.E., J.P., F.R.Ae.S., President of the Manchester Branch: Although this lecture by Harold Rogerson has been a little nostalgic for many of us I still enjoyed it and I have had little chortles to myself on some of the remarks and many of those photographs that he has shown brought back memories that go too far back.

I have very pleasant recollections of Roy Chadwick which spread over some 35 or more years and I think that, although we had a row about every other day, we were always very firm friends. We would row together but would not let anybody else row about us.

You have heard all about Roy Chadwick's achievements but I was just thinking of what the lecturer said about transonic and supersonic aeroplanes calling for a new type of designer. They do not call for a different sort of designer than Roy Chadwick, because I always found that as times changed, and we saw a lot of changes between us, Chadwick changed ahead of them. He did a lot of things that have been perpetuated by other people since. He went into rotary wing aircraft and a lot of other things too and he did not go out of them because he lost faith; he went out of rotary aircraft simply because the people with whom he was working were so stupid technically, in his view, that he could not persuade them to do the things that he knew would put the job right—most of which have since been done.

Well, that was Chadwick. I do not think you need a different type of man to design our present-day aeroplanes because most of our successful designers have come up through the mill. Our younger designers today, such as Roy Ewens, are perhaps not a new breed, they are just a later breed, slightly younger and, I think, compared with some of we old chaps, they have, perhaps, brighter ideas and are a little more bold in putting

forward their schemes and bringing them to fruition. I think we must all agree that Harold Rogerson has painted the picture of Roy Chadwick very well indeed. I agree with you, Harold, that Chadwick will go down as one of England's great designers—indeed one of the world's greatest designers—and I am delighted to know that our Branch of the Royal Aeronautical Society has taken this step in perpetuating his memory.

SIR WILLIAM FARREN: What we have heard of Roy Chadwick's achievements is enough to make us all realise that he was one of the great designers of aircraft, not only of this country, but of all countries.

I know he would have been the first to say that he did not do it all himself. He worked with a fine team of men, many of whom I am glad to say are still with us. Many others have come back for this occasion. Roy Chadwick was a magnificent team leader, and a great friend of mine. I enjoyed coming here in the old days because he cheered me up. Indeed, I think he cheered everybody up. He had that touch. I never saw him without his smile. I think one of the reasons why we liked each other was that we shared what animates all people who do anything worth doing with aeroplanes, a love of flying for its own sake.

To come down to my own association with the work Chadwick started and left only too soon, I can say that I have never ceased to remember that he was well aware that we should be undertaking a great venture, with a kind of aeroplane which, while not unique, in the sense that no one had ever thought of such a thing before, had never been brought to a successful conclusion. I think one of the things that would have pleased him most would be to know that we have now made a useful aeroplane out of it, and that we recognise that he started it, not alone, but with the team he had built up, who are as pround of the part they have played as he would have been of them.

I have recently realised that I did not know Roy Chadwick until 1928 when I came to Manchester, not for the first time, but for the first time as a visitor to A. V. Roe's, when Mr. J. D. Siddeley (later Lord Kenilworth) became interested in the Company. I am not sure that I was entirely welcome. I do not mean personally. But being taken over by anybody was not exactly the kind of thing that Sir Roy Dobson or Roy Chadwick really yearned for. Nevertheless they were polite to me, and I can say that Roy Chadwick and I liked one another from the start, and after that I came whenever I could find a reason. I remember coming to give a lecture here. I cannot remember what it was about, but I can remember that Roy took me out to an early supper to a place which I believe has now disappeared owing to the activities of the Germans. It was somewhere not far from here, and I well remember an incident, when we had settled down in the comparatively empty dining room and were talking; suddenly someone at a nearby table threw a little ball made of some fluffy stuff at us; I was a bit surprised, and disregarded it, but then another one came and Roy looked up and said "We have hit on a gala night, Bill. This is night-life in Manchester." We joined in the fun, which was not fast and furious—there were not enough people there—but I always remember the way Roy enjoyed my surprise, and I think we did our bit in keeping up the other people's spirits.

I had a good deal to do with Roy Chadwick during the war. I then came in an official capacity, but again was received very kindly; we had disputes about things which were not going quite as quickly, or not quite as well as we had expected. But I never remember us quarrelling. Roy used to drive me home, driving over the slippery setts with which Manchester streets were then paved, at a pace which frightened me. He never seemed to take the slightest notice of that, and I gradually lost my nervousness. I remember one night when the sky in the distance was lit with the flashes of bombs, but that did not seem to disturb him either.

I am happy that it was my privilege to work so closely with Chadwick, and today I feel even more what a privilege it has been over the past eight years, using his own desk and his own chair in his own room, to help to finish off the work which he began.

MR. N. E. ROWE: It only remains for me now to say a few words about the lecturer. I think it would be quite inappropriate, much as I would like to, to talk to the Branch as a Branch—but I will come back again perhaps some other time and say a few words on that subject.

The ocasion has been splendidly organised and for that I would offer your best thanks to the Chairman of the Branch, Mr. Fielding, who has done so much work for it, and Mr. Waterfall and the Committee who have, I know, put a great deal of time into this. Mr. Waterfall has been Secretary of the Branch for many years and any Branch knows how much it owes to its secretary.

We have heard about a peer of aeronautics. The record will be historic. This Branch, as Sir Roy Dobson has said, has given itself a privilege in founding this lecture which will commemorate the memory of this great man. We all, I am sure, feel as I do, deeply grateful and thankful to Mr. Rogerson who has gathered these facts together and presented them in such a homely, such a modest and such a friendly way. He has recaptured the atmosphere, the personality of a man. He has put something before us in a way which I am sure, to those who knew him, has brought Roy Chadwick vividly before us again.

Propellers for Military and Civil Aircraft

by

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Introduction

The main purpose of this paper is to outline the position which the propeller holds in current aircraft propulsion, its construction, operation and its future.

The paper has been arranged in four parts, as follows:—

1. Present and future types of aircraft foreseen with propellers.
2. Engines for turbo-prop power units.
3. The Propeller.
4. The Future.

1. Present and Future Types of Aircraft Foreseen with Propellers

In the *present military* field propellers are used for the following types of aircraft:—Heavy transport and Army reconnaissance, for example the Beverley and Pioneer, Submarine reconnaissance, Naval Strike Fighters and R.A.F. Trainers as exemplified by the Gannet, Wyvern and Provost.

The *civil field* covers twin- and four-engined feeder aircraft, four-engined trans-continental and four-engined trans-oceanic aircraft. The types are represented by the Herald, Friendship, Viscount, Vanguard and Britannia aircraft. The all-up-weight range covered by these aircraft is 30,000 to 160,000 lb.

Looking to the future, say five years ahead, the *military* applications will only disappear in respect of the carrier-borne Strike Fighter, which will become a straight jet and possibly a straight jet R.A.F. Trainer, although even here the propeller Trainer may continue in parallel.

In the *civil field* the twin-engined feeder aircraft and the four-engined trans-continental aircraft up to 100,000 lb. weight, the latter cruising at 400 m.p.h. at 30,000 ft., will still figure prominently in world-wide operation with a possible addition within this range of a twin-engined turbo-prop aircraft of around 50,000-60,000 lb. A.U.W. A still heavier aircraft of 200,000 lb. cruising at around 500 m.p.h. at 40,000 ft., will probably be the largest size to use turbo-prop propulsion.

2. Engines for Turbo-Prop Power Units

There is little doubt that piston engine development has called a halt at 3,500 b.h.p., and even this rating is a development of existing types; below this power it is certain that, from 1,000 b.h.p. upwards, there will be no new piston designs. Thus, over the entire power range, the field is left to the turbo-prop.

Existing turbo-prop units under active development range in take-off power from 1,000 to 5,000 s.h.p. Three types of engine construction have emerged, the "direct connected," the free turbine and the two-spool configuration. In the aircraft types already reviewed there is no doubt that, with the exception of the smallest of trainers and reconnaissance types, all will be powered by engines within this range.

For the future we shall see possibly further developments of existing engines and new designs extending the range to 12,000 s.h.p.

3. The Propeller

3.1. BLADE DESIGN AND CONSTRUCTION

Two forms of blade construction generally apply at the present day, *i.e. Solid Duralumin* and *Hollow Steel.*

Taking first *Solid Duralumin*, this form of construction has been used for at least 25 years, being originally fitted to adjustable pitch propellers in the U.S.A. and later finding universal use in all V.P. propellers except for the war-time expediency of wooden blades. The physical properties of the alloy have been 24 tons/in.^{2} ultimate strength with 13·5 tons/in.^{2} 0·1 per cent. proof, the latter figure being the basis on which acceptable stress levels are determined. Higher strength alloys physical propertiesof the alloy have been 24 tons/in.^{2} proof, but recently abandoned on the grounds of inconsistent fatigue properties.

The extreme manufacturing flexibility of the duralumin blade has enabled it to keep pace with the ever-increasing engine powers and aircraft performance, by virtue of the relative ease in changing the design of aerofoil section and the plan form shape as the demands arose.

Duralumin has been, and will continue to be, the predominant material for propeller blades as long as they are used for aircraft propulsion. By virtue of its "solid" construction, the limit of its practical use occurs when aircraft performance demands the use of extremely wide chord blades. At the extremely wide chords the centrifugal twisting moments assume such high values as to make the conventional form of hydraulic propeller impracticable. This is largely because the operating pressure, limited by that permissible through the engine reduction transfer muff, demands a prohibitive size of control cylinder and attendant accessories.

The practical maximum size of a duralumin-bladed propeller would appear to be 15 ft. diameter × 16 in.

A Section Lecture given before the Society on 9th February 1956.

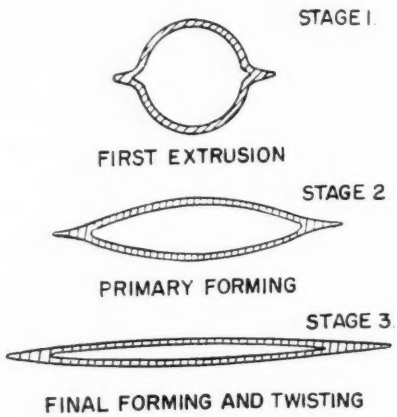

FIGURE 1.

chord or 16 ft. diameter × 15 in. chord. In this range and beyond, the *hollow construction* takes over and, at the present state of the art, the only practical form is hollow steel.

Thus far we have only considered the hollow steel at the point where it becomes essential from structural and control reasons. It should be added that this form of blade has been used in the U.S.A. for many years past in direct competition with duralumin, affording longer service life by virtue of its greater resistance to stone damage. This feature is all the more important where, for performance reasons, the blade thicknesses have to be the very minimum possible.

For the future, both forms of construction will no doubt proceed in parallel with an increased use of hollow steel as cruising speeds exceed 400 knots. Further development will no doubt extend into some form of hollow duralumin blade. The only practical method which the author can see of producing such a blade is by the extrusion process from a solid billet to tubing and thence forward through "flattening" processes to the finished blade.

It would appear to be essential that such a blade must have "integral ears" formed on the tubing during the extrusion process (as done by Curtiss Wright in their hollow steel extruded blades), to ensure the sharpness and fineness of the leading and trailing edges in the final finished form of blade (*see* Fig. 1). Just how duralumin tube will lend itself to the 'flattening' process is questionable and will require some considerable development.

3.2. HUB AND ACCESSORY DESIGN AND CONSTRUCTION

The conventional construction comprises the propeller itself, shaft-mounted to the engine with the control accessories mounted on the engine, as shown in Fig. 2. The oil supply for propeller control is taken from the main engine system, boosted in pressure within the C.S.U. and then fed through an engine-mounted transfer muff to the propeller control cylinder.

The blade bearing mounting in the hub socket, in addition to its basic duty of handling the centrifugal and bending loads, plays an important part in the degree of damping afforded in the blade vibratory system. Futhermore, on account of the extreme importance of good aerofoil shape at the spinner periphery on turbine engines, particularly with annular air intakes, the blade mounting design must be of a squat nature to ensure that the hub sockets are completely contained within the spinner.

Propeller pitch control, normally responsive during constant-speeding to an overspeed or underspeed signal from the C.S.U. governor, can also be made responsive to an acceleration or deceleration signal, an advantage with high inertia engine compressors on direct coupled turbine engines.

Manual and automatic feathering, along with automatic propeller synchronising have been featured for some considerable time. The auto-synchronising feature however is now being extended to phase-synchronising wherein the angular position of the blades is maintained in a predetermined position relative to each other, *e.g.* in the case of inboard

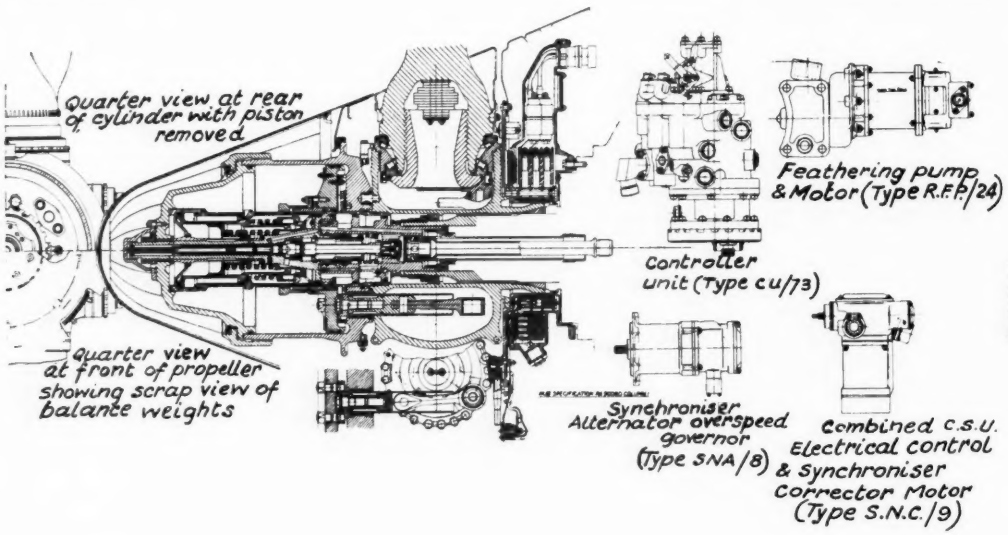
Quarter view at rear of cylinder with piston removed
Feathering pump & Motor (Type R.F.P./24)
Controller unit (Type C.U./73)
Quarter view at front of propeller showing scrap view of balance weights
Synchroniser Alternator overspeed governor (Type S.N.A./8)
Combined C.S.U. Electrical control & Synchroniser Corrector Motor (Type S.N.C./9)

FIGURE 2. 16 ft. diameter dural blade on No. 6 S.B.A.C. propeller shaft.

propellers the blade relation is such that they consistently pass the fuselage at a predetermined relation to each other. This feature has been shown to reduce considerably the noise level.

Propeller reversing has also been featured for some time past, although further developments are proceeding in the provision of a manual "inching" control of propeller pitch after touch-down, which ensures a smooth slowing down of the aircraft during the early part of the landing run with the normal reversing still available if circumstances necessitate. This feature is also of considerable assistance during taxying.

3.3. CONTROL AND SAFETY FEATURES

The "line diagram" shown in Fig. 3 indicates very clearly the various propeller control and safety features and traces the steps taken in the system to cater for as many failures as is practicable.

The safety features shown afford protection against:—

(*i*) Engine failure.
(*ii*) Loss of primary oil supply to propeller controller unit.
(*iii*) Failure of drive to propeller controller unit.
(*iv*) Controller unit relief valve jammed in open position.
(*v*) Constant speed unit valve jammed in fine pitch position.
(*vi*) Loss of oil pressure at engine transfer.
(*vii*) Failure of main oil seals within propeller which affect control.

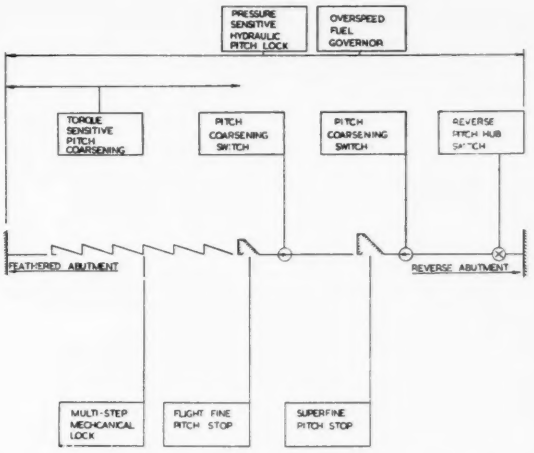

FIGURE 3. Line diagram of safety features.

These safety features, taken in order from the reverse abutment, are as follows:—

Reverse pitch hub switch. Operates lock in throttle box that will prevent reverse movement of throttles beyond reverse idling until blades reach reverse abutment.

Pitch coarsening switch. Acts as an unbraking switch. Also as a secondary protection 2° behind superfine pitch stop.

Superfine pitch stop. Removable mechanical lock

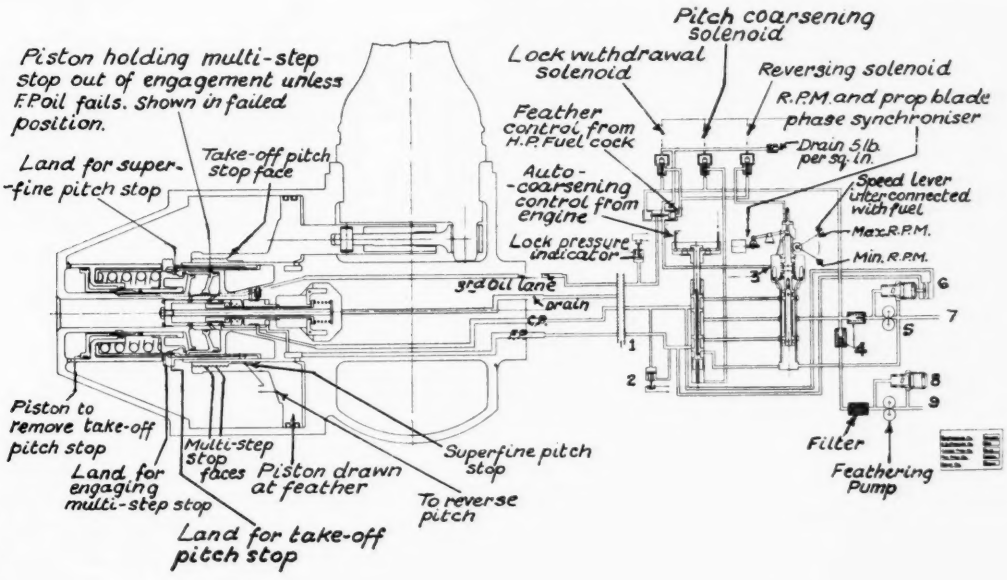
Pitch coarsening solenoid
Piston holding multi-step stop out of engagement unless F.P. oil fails. Shown in failed position.
Lock withdrawal solenoid
Reversing solenoid
R.P.M. and prop blade phase synchroniser
Feather control from H.P. Fuel cock
Drain 5 lb. per sq. in.
Land for super-fine pitch stop
Take-off pitch stop face
Auto-coarsening control from engine
Speed lever interconnected with fuel
Lock pressure indicator
Max. R.P.M.
Min. R.P.M.
3rd oil lane
drain
Piston to remove take-off pitch stop
Multi-step stop faces
Land for engaging multi-step stop
Piston drawn at feather
Superfine pitch stop
To reverse pitch
Land for take-off pitch stop
Filter
Feathering Pump

FIGURE 4. Propeller hydraulic control system (diagrammatic only).

1. Engine transfer.
2. Pressure cut-out switch.
3. Bleed 3rd oil way transfer lubrication.
4. Non-return valves.
5. Governor Pump
6. Compensated relief valve & emergency valve
7. Feed from engine 50 lb. per sq. in.
8. Relief valve
9. Oil supply from tank.

CASE 1

AIRCRAFT DATA

4000 s.h.p. for take-off
2600 s.h.p. for cruise at 425 m.p.h. at 25,000 ft.

PROPELLER DATA

4 blades, 15 ft. diameter, 16 in. max. chord
Solid duralumin
N.A.C.A. series 16 sections

Integrated design C_L	=0·55
Thickness chord ratio at tip	=3·75 per cent.
" " " at 0·7 radius	=6·7 per cent.
" " " at spinner	=35 per cent.
Activity factor	=136
Take-off thrust	=2·43 lb./s.h.p. static
" "	=2·205 lb./s.h.p. 100 m.p.h.
Free air cruise efficiency	=86 per cent.

CASE 2

AIRCRAFT DATA

5000 s.h.p. for take-off
3750 s.h.p. for cruise at 495 m.p.h. at 35,000 ft.

PROPELLER DATA

4 blades, 16 ft. diameter, 18 in. max. chord
Hollow steel construction
N.A.C.A. series 16 sections

Integrated design C_L	=0·3
Thickness chord ratio at tip	=2·6 per cent.
" " " at 0·7 radius	=5·5 per cent.
" " " at spinner	=35 per cent.
Activity factor	=145
Take-off thrust	=2·28 lb./s.h.p. static
" "	=2·10 lb./sh.p. 100 m.p.h.
Free air cruise efficiency	=80·5 per cent.

set as required for ground running, taxying and windmill braking. The stop is removed by third oil line pressure when operation in reverse pitch is required.

Pitch coarsening switch. As a secondary protection 2° behind the mechanical fine pitch lock. Also indicates blades in low pitch constant speeding range.

Flight fine pitch stop. Lock removed for ground running, windmill and reverse pitch braking by third oil line pressure.

Multi-step mechanical lock. Above flight fine pitch stop the multi-step mechanical stop locks blade angle at setting within 4° of pitch at instant of failure.

N.B. The lock design still permits coarsening of pitch for feathering when required.

Pressure sensitive hydraulic pitch lock. Operates upon loss of fine pitch oil pressure. Closes the coarse pitch oil line with a non-return valve and thus prevents the pitch moving fine.

N.B. The design still permits coarsening of pitch for feathering when required.

Overspeed fuel governor. The engine incorporates an overspeed progressive fuel governor set to operate at 10 per cent. to 12 per cent. above normal maximum r.p.m.

Torque sensitive pitch coarsening. Hydraulic type torque meter incorporated in the reduction gear actuates pitch coarsening system when torque reaches negative value. Caters for an engine failure on take-off or during cruise.

Having now seen the construction of the propeller in its physical sense, Fig. 4 shows a typical diagrammatic circuit of the propeller.

3.4. AERODYNAMICS

From the aerodynamic aspect, the main interest lies in the attainable propeller cruising efficiencies at the higher forward speeds while maintaining a good take-off thrust. Instead of generalising, it is better to confine the aerodynamic problems to typical aircraft. For this purpose the aircraft characteristics under consideration are:—

Case I. 4,000 s.h.p. for take-off.
2,600 s.h.p. for cruise at 425 m.p.h. at 25,000 ft. and a still more advanced design with:—

Case II. 5,000 s.h.p. for take-off.
3,750 s.h.p. for cruise at 495 m.p.h. at 35,000 ft.

In the first case the aircraft could be propellered with the conventional duralumin-bladed type, as the cruising performance is not jeopardised by the use of high camber aerofoil sections which would be necessary to meet take-off requirements without the use of unduly wide blades. Such a propeller would consist of:—

4-blades × 15 ft. diameter × 16 in. max. chord.
Integrated design C_L = 0·55.
N.A.C.A. 16 sections.
Thickness chord ratio at tip = 3·75 per cent., at 0·7R = 6·7 per cent., spinner = 35 per cent.
Activity factor = 136.
Take-off thrust 2·43 lb./s.h.p. static.
2·2 lb./s.h.p. 100 m.p.h.
Cruise efficiency = 86 per cent. free air.

In the second case, the aircraft cruising speed and power is such that use of a high camber blade to achieve satisfactory take-off and climb would drop several per cent. on cruise efficiency. A propeller for this application would comprise:—

4-blades × 16 ft. diameter × 18 in. max. chord.

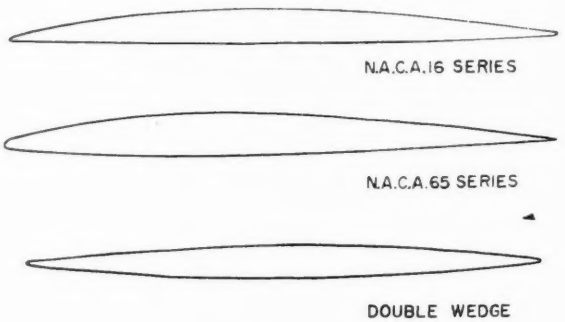

FIGURE 5.

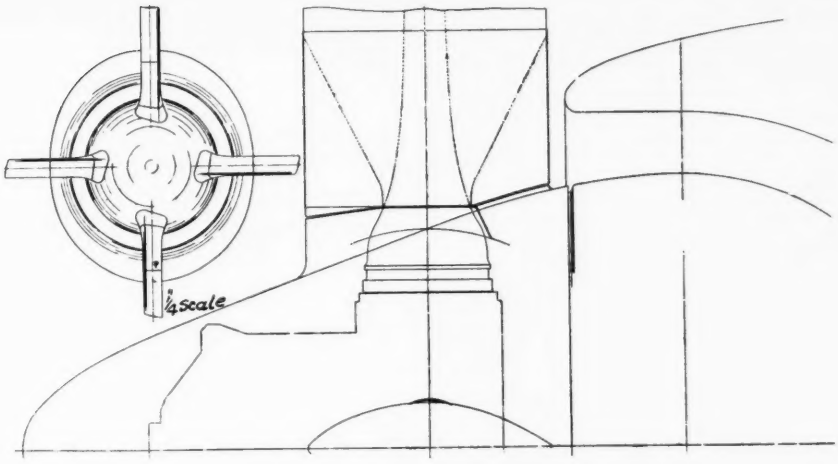

FIGURE 6. Fixed blade root spinner fairing.

Hollow steel construction.
Integrated design C_L of 0·3.
N.A.C.A. 16 sections.
Activity factor 145.
Take-off thrust 2·28 lb./s.h.p. static.
2·1 lb./s.h.p. 100 m.p.h.
Cruise efficiency 80·5 per cent. free air.

Somewhere in between these two aircraft lies a configuration at which the aerodynamic performance cannot be met with a practicable design of solid duralumin blade. In both Cases I and II however, the extreme importance of eliminating any external excresences on the blade aerofoil cannot be over-emphasised. Also of equal importance is the production of a true aerofoil section demanding no mean amount of careful consideration in the case of the hollow steel construction. Attendant to the hollow steel blade in Case II is a self-contained hub construction to operate these blades, which will be referred to later.

At present, the best known aerofoil section is the N.A.C.A. series 16, although there may be a swing over to N.A.C.A. series 65 on inboard sections to give improved take-off thrusts (*see* Fig. 5).

In 1945 an experimental propeller with faired double wedge sections was tried on a Supermarine Spiteful aircraft and gave 498 m.p.h., which was a little faster than that given by an identical series 16 sectioned propeller. Unfortunately the propeller was written off in a landing mishap and Supermarines, being by that time in the jet age, had no further application of the development. While the section was very good for high speed, it suffered under take-off and climb conditions, due to early stall characteristics.

Within the scope of the aerodynamic side of the propeller comes the extreme care required in the design of the blade at its junction with the spinner, particularly with the gas turbine engine wherein the efficiency of the air flow over the blade roots can have a marked effect on engine intake conditions. Of various developments carrid out thus far, it appears that a fixed spinner fairing, shown in Fig. 6, mating with the blade root aerofoil at cruising angle, is the best compromise.

3.5. PROPELLER REVERSE THRUST

In the case of the landing brake, *i.e.* reverse thrust from the propellers, the advantage afforded by its supplementing the wheel brakes and also for icy runway operation, is already fully known. The principle of applying reverse thrust follows closely the piston engine practice wherein a fixed reverse angle is reached and then the engine throttles re-opened. It was at one time thought that the direct connected engine would demand the constant speed control being extended into the reverse pitch range, but this has since been considered to be an unnecessary complication.

It is possible to obtain a considerable braking effect from the propellers by allowing them to windmill at low blade angles (for example 0°) immediately after touch-down. Almost all turbo-propellers use this form of braking and future developments will incorporate this feature together with power-on braking.

3.6. PROPELLER SHAFT BRAKE.

With the gas turbine engine, and particularly forward loading of passengers, it has become necessary to provide a propeller shaft brake to prevent the propeller windmilling with the aircraft static while loading passengers. It is also necessary in order to reduce the run-down times of the engine/propeller combination.

The propeller shaft brake is as yet in its infancy and is being fitted wherever most convenient. Initially all new engine/propeller installations will have to cater for this requirement in the most advantageous position to reduce bulk and weight.

3.7. DE-ICING

Propeller de-icing is now almost 100 per cent. electric and, in principle, consists of a rubber shoe fixed along the leading edge of the blade and heated from the aircraft electrical system through slip rings behind the propeller hub.

In view of the high electrical load associated with continuous heating, it has been found essential to cycle load for the blade, spinner and engine intake.

To prevent any spoiling of the aerofoil section, the

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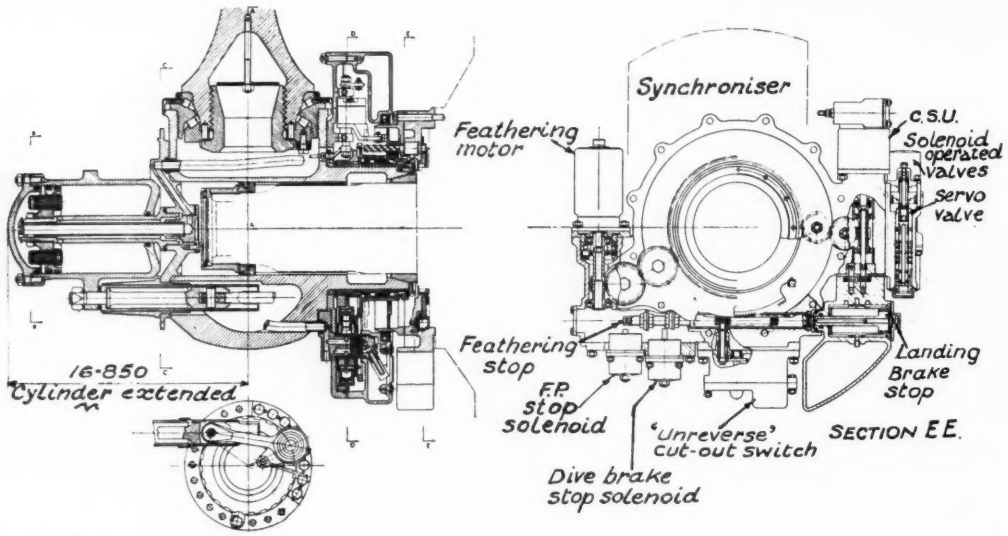

FIGURE 7. Self-contained hub construction.

de-icing element is rebated into the leading edge of the duralumin blade, thus preserving a flush surface over the aerofoil. In the hollow steel form of blade, efforts have been made to carry the element internally and so avoid damage by stones, etc., but so far without any practical success, as any failure of the element electrically renders the blade unserviceable.

While the only practical answer is the external element, special steps have to be taken by providing a thin stainless steel protecting sheath over the leading edge nose of the de-icing shoe. This sheath is so thin and, being partially embedded into the rubber overshoe, no loss in performance occurs. It has, however, been shown to extend the life of the shoe most appreciably.

The spinner de-icing is usually graded to give a greater intensity of heat at the nose, reducing in intensity towards the rear.

Typical de-icing requirements for a 16 ft. diameter prop. of 15 in. chord are 2·5 kw./blade and 6·0 kw./spinner with an intensity at the nose of 11 watts/in.^{2} and at the rear 8·5 watts/ in.^{2} The cycling arrangement is usually 30 seconds on and 90 seconds off. A second speed is usually available keeping the ratio the same.

3.8. VIBRATION FATIGUE PROBLEMS

Structural fatigue problems have been with the propeller and engine designer for 15 to 20 years and are not, as one might be led to think, a recent phenomenon in aircraft design.

In fact, the very creation and development of strain gauging was done in the U.S.A. on propeller development and afterwards applied to all forms of engineering. From the very start the propeller designer, allied with his colleague the aero-engine designer, has been most conscious of stress concentration and has given very careful study to the most detailed elements of the design to avoid potential stress raisers.

In piston-engined propellers the predominant exciting forces arise from the reciprocating and crankshaft system and, during the era of engine ratings between 2,000 and 3,500 b.h.p., have to a great extent dictated the structural strength of the propeller, the pure aerodynamic performance requirements suffering in consequence. Due to the absence of these engine excitations, turbo-propeller weights have been reduced and in consequence the blades are more susceptible to aerodynamic excitations. One of the main excitations is the first propeller order.

Recent mathematical studies of the first propeller vibration problem have produced an accurate method of estimating the vibratory stresses arising from these excitations during the initial design of the blade. At the same time, the propeller designer is now sufficiently knowledgable of the problem to advise the aircraft designer of the optimum nacelle angle relative to the "no-lift" angle of the wing for minimum first propeller order.

In addition to the pure theoretical determination, strain gauging of a new propeller type or blade is done on the engine hangar, followed later by a further strain gauging in flight before the final clearance is given.

Flight strain gauge results are first used to establish a reasonable fatigue life of the blade. After that the blades are returned from Service at say 2,000 hour intervals, and subjected to a vibration fatigue test to establish whether deterioration has taken place or not. This procedure enables the rig results to be always in advance of actual flying hours and also enables, over a period of time, the ultimate life of the blade to be determined. A similar exercise is applied to the blade root mounting in the hub socket.

3.9. NOISE

With the advent of the relatively quieter turbine engine, accent has been placed on reducing propeller noise.

It becomes necessary, therefore, in modern propeller design to consider the noise level at the very outset and determine the propeller r.p.m. at the cruising condition with this in mind. The situation is made more difficult

by the " nearness " of take-off and cruise r.p.m. on turbine engines, which tends to give a poor take-off thrust if the gear ratio is fixed at too low a value, although recent engine developments on ' up-speeding ' for take-off have considerably helped in keeping down the noise level at cruise.

Other factors, such as proximity of blade tips to the fuselage, play an important part in the noise level. Whether the latest practice of square tips adjacent to the fuselage on the inboard engines are worse than rounded tips, is very questionable, as tests carried out thus far indicate no appreciable reduction in noise, but a certain amount of reduction in the vibration level.

The effect of power loading per blade is also a factor in the noise level and has to be taken into consideration during the design. Increase of number of blades leads to a reduction in noise level, but the use of more than four blades in a propeller appears at present unacceptable to operators, although it is considered quite practicable to build five-bladed propellers with duralumin blades.

Tests conducted in U.S.A. have indicated a marked reduction in noise level by " phase synchronising " the propellers. The author would prefer to reserve his views on the extent of this reduction until parallel tests at present underway in this country are completed.

3.10. COUNTER-ROTATING PROPELLERS

The counter-rotating propeller, with either the " double-three " or the " double-four " arrangement of blades, enjoyed considerable success both during the last war and subsequently on Fighter and Bomber aircraft.

This type of propeller, however, has never found favour on civil aircraft, largely on account of its greater expense, complication and more complex service and maintenance problems.

At the same time, however, it must be pointed out that by and large, the propeller diameter afforded on civil aircraft has enabled the single hub with a maximum of four blades to be satisfactory thus far, although at certain restricted diameters the 5-blader has been looked at, but each time turned down on initial and subsequent maintenance expense. It does not appear likely that the counter-rotating propeller will ever appear on future civil aircraft; the 5-blader may, but it is very doubtful.

The co-axial arrangement, *i.e.* each half driven by an independent engine, is still favoured on the Naval reconnaissance aircraft, *e.g.* Fairey Gannet, and will continue as long as this form of aircraft uses propeller propulsion.

4. The Future

4.1. AERODYNAMICS

So far a forward speed of 500 m.p.h. at 40,000 ft. has been discussed and it has been shown that propeller free air efficiencies of the order of 80 per cent. are attainable.

The problem now arises as to whether such efficiencies can be achieved at still higher speeds, up to say 600 m.p.h. It is questionable whether a " *Transonic Propeller,*" *i.e.* subsonic for the greater length of the blade but supersonic at the tips, will still be an efficient proposition at 600 m.p.h. As far as aerodynamic data takes us it appears that the " transonic propeller " would fall to 60 per cent. efficiency at 600 m.p.h. mainly on account of the high induced losses. If, however, the r.p.m. be increased at the expense of higher compressibility losses, the gain in induced efficiency is such as to raise the overall efficiency to the order of 68 per cent. If the hollow steel blade could be designed structurally to give Tm/c gradings of 2 per cent. at the tip, 4 per cent. at 0·7*R* and 12 per cent. at the spinner, a recovery of some 3 to 4 per cent. in efficiency is possible, which brings it comfortably back into the picture, although this propeller is now more nearly a supersonic propeller.

The next consideration is the purely " Supersonic Propeller " wherein the rotational and forward speed combine to give supersonic conditions over the whole

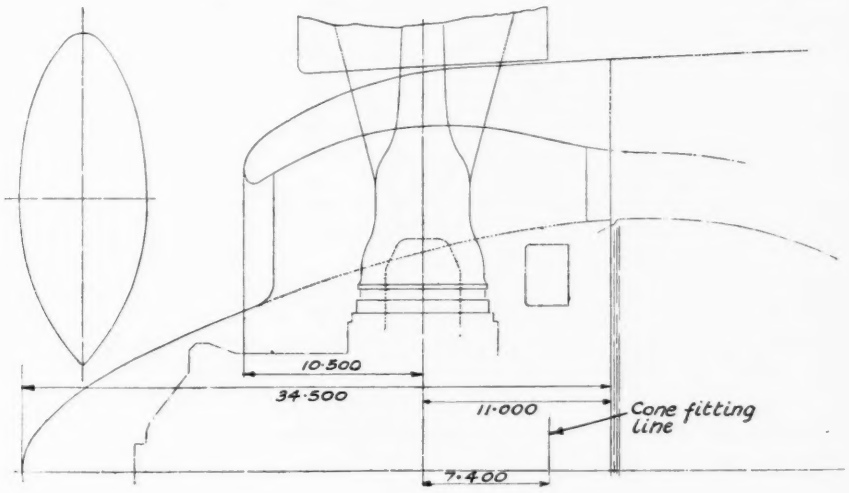

FIGURE 8. Contour of ducted spinner.

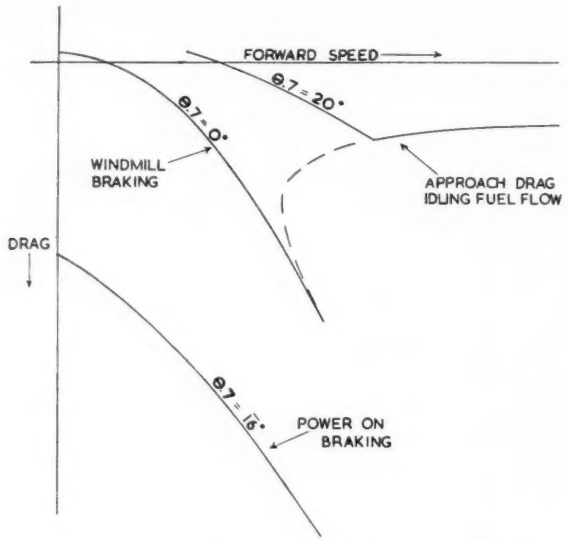

FIGURE 9. Landing drag curve.

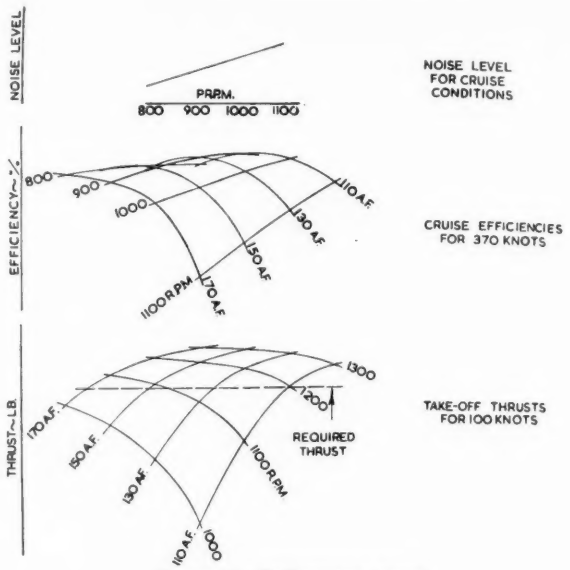

FIGURE 10. Performance carpet.

length of the blade. Here again, however, blades of still further extreme thinness ratio are necessary and may only be practical in solid steel. Aerodynamic data to hand, albeit somewhat sparse, suggests that free air efficiencies of 70 to 76 per cent. are possible at 600 m.p.h. This is accounted for by a reduction of aerofoil drag/lift ratio beyond a Mach number of approximately 1 – 1·1 and also by an increase in induced efficiency through working at a more favourable advance ratio (*i.e.* ratio of forward to rotational speed).

The most serious obstruction to the supersonic propeller is in its extremely high noise level, which at the present time would be entirely out of the question, not only on the grounds of passenger comfort, but also from the aspect of general public nuisance, particularly in the light of recent agitations in this direction. This high noise level could in itself be the death-knell of this form of propeller.

In general though, the governing factor as to whether propeller-driven aircraft will ever be called for at 600 m.p.h. rests with the success or otherwise of the straight jet civil air liners, once they have had operational experience. The author has little doubt that such problems as providing some practical form of reverse jet thrust and also lower engine fuel consumption will be conquered, leaving the field beyond 500 m.p.h. to this form of aircraft. This does not infer that the jet aircraft will oust the 500 m.p.h. turbo-prop.; rather both will proceed in parallel within their respective spheres.

Nevertheless, intensive development is essential on the structural side of the blade design to achieve thickness/chord gradings commensurate with the forward speeds in question.

4.2. HUB CONSTRUCTION

While dealing with the future, the hollow steel blade required to meet the performance in Case II (*see* Section 3.4) will, because of its wide chord and extremely high centrifugal loads and high centrifugal twisting moments, demand a form of self-contained hub construction wherein the complete propeller is divorced from the engine, having its own oil reservoir and driving and mounting its accessories. Such a construction, as shown in Fig 7, enables the propeller designer to use higher operating pressures to control the wide blades, being no longer hampered by the engine transfer muff limitation, and also to select the best all-round fluid without having to consider the implications of the engine.

This form of hub construction is not by any means new in the U.S.A., but has still to be established in the British propeller industry.

4.3. DUCTED SPINNERS

Considerable development work has been carried out on the ducted form of spinner indicating improved intake performance thereby (*see* Fig. 8). Thus far, however, no operational aircraft has been equipped with such spinners, largely on account of the heavy and extremely complicated structure and also the problem of de-icing for which, while many ingenious ideas have been forthcoming, no practical solution has yet been achieved. The increased weight and complexity of a ducted spinner negatives the increase in intake efficiency.

It may be however, that for speeds above 500 m.p.h. such spinners will be aerodynamically essential, in which case the added weight and complication would have to be offset against the gain in performance.

The Aerodynamics of the Helicopter in Forward Flight

by

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Introduction

Existing methods of estimating the blade loadings on a helicopter rotor in forward flight have suffered from the necessity of using false simplifying assumptions in order to make possible an analytical approach. The following list gives the most usual of these assumptions: —

(*i*) Blades have no twist or taper, or are twisted and tapered in a particular way.

(*ii*) The coning angle is small enough to neglect the resulting changes in velocities, angles, and so on.

(*iii*) The disc is horizontal.

(*iv*) The components of velocity parallel to the blade length may be neglected, and forces are calculated for and act with reference to the velocities normal to the blade length.

(*v*) There is no variation in downwash over the disc, or the variation is capable of simple expression.

(*vi*) The blades extend from hub, and tip losses are neglected.

(*vii*) The blades have pure flapping hinges and the analogy between flapping and feathering enables loadings calculated for flapping blades to be referred to a "hingeless" rotor system.

Examination of these assumptions and an appreciation of the problem lead to the conclusion that a calculation by the method of elements (strip theory) gives the only reliable solution; and the enhanced accuracy of the result justifies the considerable increase in labour that is incurred.

For a flapping blade rotor system uncertainty concerning the accuracy of the blade loading calculations is not necessarily serious. When a hingeless rotor system is employed accuracy is desirable, if not essential, because of the need to be certain of the fatigue strength of the blade roots.

The strip theory method can give also a solution of the precise blade feathering motion required to balance in any condition of flight. If the whole of the rotor control is vested in blade feathering—as with a hingeless rotor—then the use of a non-sinusoidal feathering motion derived by this solution gives rise to the hope that not only will fatigue stresses at the blade root be a minimum but also the overall rotor vibrations associated with loaded rotors at high advance ratios.

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NOTATION

Forces:

D_i	drag on blade induced by lift vector
D_0	two-dimensional drag of blade profile at zero lift coefficient
D_0'	two-dimensional drag of blade profile at non-zero lift coefficient
D_P	resultant parasitic drag, *i.e.* drag extraneous to main rotor
D_{PH}	parasitic drag in flow parallel to rotor disc plane
D_{PV}	parasitic drag in flow normal to rotor disc plane
H_i	force in plane of disc due to components of D_i in line of flight
H_0	force in plane of disc due to components of D_0 in line of flight
$H=H_0+H_i$	
L	lift vector on blade normal to incident flow
l	rolling moment about rotor hub due to forces external to rotor
l_a	rolling moment about rotor hub due to forces within rotor
$l_{C.F.}$	rolling moment due to centrifugal forces arising from the variable component of the thrust loading
M	pitching moment about rotor hub due to external forces
M_a	pitching moment about rotor hub due to forces within rotor
$M_{C.F.}$	pitching moment due to centrifugal forces arising from the variable component of the thrust loading
N_i	component of L normal to blade length and tangential to disc cone at blade
N_0	component of D_0' normal to blade length and tangential to disc cone at blade
Q_i	torque induced by lift force in blade
Q_0	torque induced by profile drag on blade
R	resultant force due to incidence acting on blade
T	thrust force acting normal to blade and in plane containing rotor shaft
T'	component of T parallel to rotor shaft, *i.e.* useful component of thrust
T''	thrust force defined by $T \sin \omega = T_{TR}$
T_{TR}	side force required for torque reaction
W	weight of helicopter

Dimensions:

x_w, y_w, z_w arms of weight-moment about rotor hub
x_D, y_D, z_D arms of parasitic drag moment about rotor hub
x_t, y_t, z_t arms of torque-reaction about rotor hub
r radial dimension from hub
R rotor radius
c local blade chord
$\bar{c}$ standard blade chord
dr element of blade $= Rdx$
S plan area of one blade

Velocities:

V incident velocity, *i.e.* that of free stream
V_a component of V parallel to rotor axis
V_c component of V normal to horizontal, *i.e.* climb velocity
V_t component of V parallel to disc plane
V_t' component of V tangential to surface of coned rotor
V_n component of V normal to surface of coned rotor
V_r vectorial sum of velocities V_t' and Ωr
V' vectorial sum of velocities V_r and $(v + V_n)$ (blade loadings)
$\overline{V'}$ resultant velocity passing through disc
V_1 level-speed, or component of V parallel to horizontal
v induced velocity

Angles:

α_r aerodynamic incidence of blade element
γ sweep angle of blade element relative to V_r
θ pitch angle of blade relative to the disc plane
θ_0 datum pitch angle, usually angle at $0{\cdot}75R$
θ' pitch angle of blade relative to V_r
ϕ_0 angle produced by $(v + V_n)$, relative to disc plane
ϕ' angle produced by $(v + V_n)$, relative to V_r
ϕ azimuth angle measured from aft
ϕ_1, ϕ_2 components of ϕ'
ψ angle between horizontal and $\overline{V'}$ (+ve measured downwards)
χ angle between disc plane and horizontal (+ve nose down)
i angle between disc plane and V (+ve nose down)
τ_c angle between V and horizontal (+ve in climb)
ω side tilt of main rotor
β coning angle for conditions considered

Coefficients:

$C_{Di} = D_i/(\frac{1}{2}\rho V'^2 CRdx) = D_i/[\frac{1}{2}\rho v^2 (\Omega R)^2 CRdx]$
$C_{D0}' = D_0'/(\frac{1}{2}\rho V'^2 CRdx) = D_0'/[\frac{1}{2}\rho v^2 (\Omega R)^2 CRdx]$
$C_{D0\gamma}'$ drag coefficient of D_0' on infinite wing at sweep angle γ
$C_{DP} = D_P/(\frac{1}{2}\rho \overline{V'}^2 \pi R^2) = D_P/[\frac{1}{2}\rho v_0^2 (\Omega R)^2 \pi R^2]$
$C_{Hi}, C_{H0} = H_i, H_0/[\pi R^2 \rho (\Omega R)^2]$
$C_H = H/[\pi R^2 \rho (\Omega R)^2]$
$C_L = L/(\frac{1}{2}\rho V'^2 CRdx)$
$\bar{C}_L = (nW/b)/[\rho S (\Omega R)^2 \times 0{\cdot}68^2]$
$C_l, C_{la}, C_{lC.F.} = l, l_a, l_{C.F.}/[\pi R^3 \rho (\Omega R)^2]$
$C_m, C_{ma}, C_{mC.F.} = M, M_a, M_{C.F.}/[\pi R^3 \rho (\Omega R)^2]$
$C_{Ni}, C_{N0} = N_i, N_0/(\frac{1}{2}\rho V'^2 CRdx)$
$C_{Qi}, C_{Q0} = Q_i, Q_0/[\pi R^3 \rho (\Omega R)^2]$
$C_R = R/(\frac{1}{2}\rho V'^2 CRdx)$
$C_T, C_T', C_T'' = T, T', T''/[\pi R^2 \rho (\Omega R)^2]$
$C_{T_{TR}} = T_{TR}/[\pi R^2 \rho (\Omega R)^2]$
$C_W = W/[\pi R^2 \rho (\Omega R)^2]$
δ average C_{D0}' for flight condition considered
C_F skin friction drag coefficient
$\mu = V \cos i/(\Omega R) = V_t/(\Omega R)$
$K = V \sin i/(\Omega R) = V_a/(\Omega R)$
$\mu' = V_t'/(\Omega R)$
$K' = V_n/(\Omega R)$
$\mu'' = V_r/(\Omega R)$
$\nu = V'/(\Omega R)$
$\nu_0 = \overline{V'}/(\Omega R)$
$\Delta = v/(\Omega R)$
$\bar{\Delta} = \bar{v}/(\Omega R) = C_T/(2\nu_0)$ by momentum theory
$\lambda = K' + \Delta$
$\lambda_0 = K + \bar{\Delta}$
F tip loss factor on thrust or lift
G tip loss factor on induced torque or drag
n acceleration on aircraft c.g.
t/c thickness-chord ratio of blade profile
$\sigma = b\bar{c}/(\pi R)$
$\sigma_r = bc/(\pi R)$
E coefficient defined in equation (21)
$x = r/R$
a_0 lift-curve slope of blade profile
$a_{0\gamma}$ lift-curve slope of blade profile on infinite wing swept at angle γ
b number of blades
B limit of integration in terms of x
K_1, K_2 coefficients in cyclic pitch terms of harmonic feathering equation

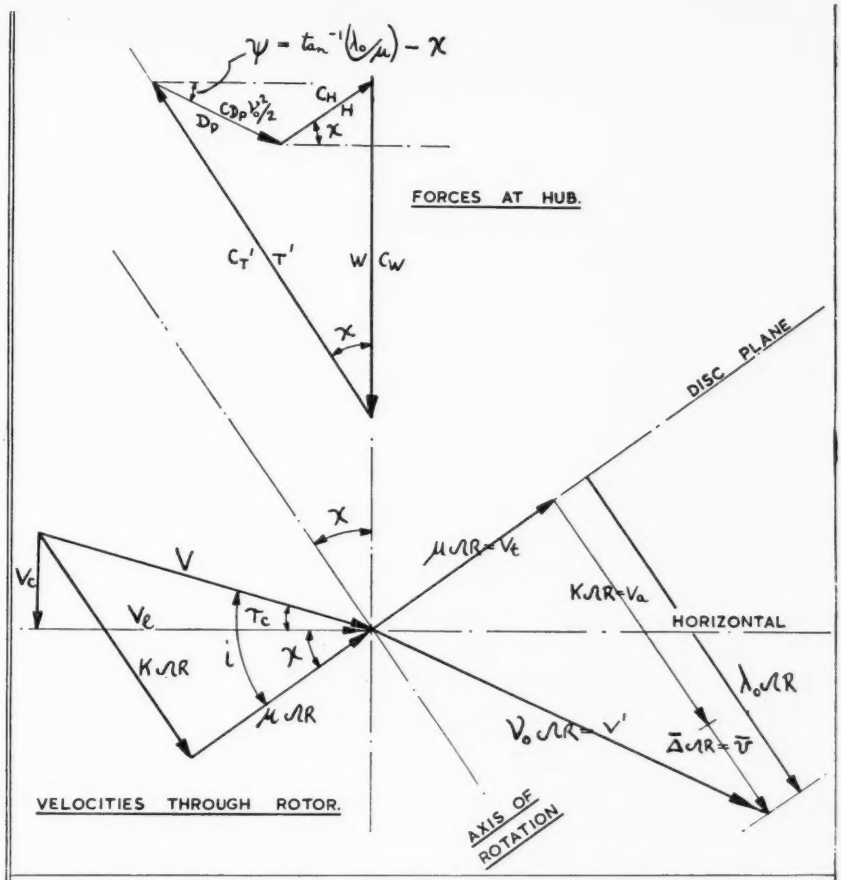
Section 1

In this section,

1.1. The rotor is assumed to have taken up a position relative to the incident flow governed by the balance of forces acting on it. Expressions for the derivation of the rotor attitude are given so that a solution of the performance equation may be made.

1.2. A refinement of the "Energy Account" method of performance estimation, conceived by Glauert[2] and further developed by Squire[1], is derived. The object of this refinement is to take account of the very large vertical drag forces which can arise when stub wings are fitted to off-load the rotor in forward flight. It tends to make the derived equations very unwieldy to use and it is suggested that it should be resorted to only when necessary.

The derived method gives the numerical solutions for the rotor attitude relative to horizontal and incident flow, enabling the components of velocity relative to the disc-plane to be obtained.



V = FREE – STREAM VELOCITY	
$V' = V_0\Omega R$ = RESULTANT VELOCITY LEAVING DISC	
$V_c = V \sin T_c$ = RATE OF CLIMB	
$V_e = V \cos T_c$ = FORWARD SPEED	
$V_a = V \sin i = K\Omega R$ = AXIAL COMPONENT OF V	
$V_t = V \cos i = \mu\Omega R$ = DISC PLANE COMPONENT OF V	
$\bar{v} = \bar{\Delta}\Omega R$ = AVERAGE INDUCED VELOCITY	
$\lambda_0 = \bar{\Delta} + K$ = NON-DIMENSIONIAL TOTAL AXIAL VELOCITY	
χ = DISC TILT TO HORIZONTAL	i = DISC INCIDENCE
$H = C_H \pi R^2 \rho (\Omega R)^2$ = FORCE IN PLANE OF DISC	
$W = C_W \pi R^2 \rho (\Omega R)^2$ = WEIGHT	
$D_p = C_{Dp} \frac{1}{2} \rho V^2 \pi R^2 (\Omega R)^2$ = PARASITE DRAG IN RESULTANT VELOCITY	
$T' = C_T' \pi R^2 \rho (\Omega R)^2$ = THRUST NORMAL TO DISC PLANE	
$\chi = \tan^{-1}[H + D_p \cos(\psi+\chi) / T' - D_p \sin(\psi+\chi)]$	

FIGURE 1. Diagram of force and velocity vectors.

1.3. The effect of the side tilt of the main rotor required to balance the torque reaction force is considered.

1.4. Parasite (*i.e.* extra-to-rotor) drag, weight and torque reaction forces do not, in general, act at the rotor hub which is used as the reference datum in this work. The moments that they produce are considered and expressions are given which are used in Section 2 to balance the helicopter in the static state considered.

1.1. THE EQUILIBRIUM OF THE ROTOR SYSTEM

The rotor disc is shown in Fig. 1 at an arbitrary angle of incidence i, exaggerated to show clearly the external forces and velocities.

In the performance equations four external forces are considered: the weight of the helicopter, the drag force of the rotor in the disc plane (sometimes called the *H*-force), and the parasitic drags of all extra-to-rotor items, in flows normal and parallel to the disc plane.

Resolving these forces parallel and normal to the disc plane gives the following balance equations: —

$$W \sin \chi = H + D_{PH} \quad . \quad . \quad . \quad (1)$$

$$W \cos \chi = T' - D_{PV} \quad . \quad . \quad . \quad (2)$$

hence

$$\tan \chi = [H + D_{PH}]/[T' - D_{PV}]. \quad . \quad (3)$$

The resultant velocity V', leaving the rotor has components parallel and normal to the disc plane of $V' \cos(\psi + \chi)$ and $V' \sin(\psi + \chi)$.

i.e. $$V'_H = V' \cos(\psi + \chi) = V'(\mu/\nu_0) \quad . \quad (4a)$$

$$V'_V = V' \sin(\psi + \chi) = V'(\lambda_0/\nu_0) \quad . \quad (4b)$$

where, from Fig. 1: —

$$\left.\begin{aligned} \psi + \chi &= \tan^{-1}(\lambda_0/\mu) \\ &= \cos^{-1}(\mu/\nu_0) \\ &= \sin^{-1}(\lambda_0/\nu_0) \end{aligned}\right\} \quad . \quad . \quad . \quad (5)$$

Expressions (1)-(3) may be made non-dimensional using coefficients as defined in the notation

$$\chi = \sin^{-1}[\sigma\delta + C_{DPH}\mu]\,\mu/2C_W \quad . \quad . \quad (6)$$

or $$\chi = \cos^{-1}[C_T' - 0{\cdot}5C_{DPV}\lambda_0^2]/C_W \quad . \quad . \quad (7)$$

or $$\chi = \tan^{-1}[\sigma\delta + C_{DPH}\mu]/[2C_T' - C_{DPV}\lambda_0^2] \quad (8)$$

In (6) and (8) the approximation obtained from Squire's work[1] is used: —

$$C_H \simeq \sigma\mu\delta/2. \quad . \quad . \quad . \quad (9)$$

Other relationships may be written down more readily:

From Fig. (1):

$$\left.\begin{aligned} \mu &= V\cos i/(\Omega R) \\ &= (\nu_0^2 - \lambda_0^2)^{0{\cdot}5} \\ &\simeq \nu_0 \cos i \\ &\simeq \nu_0 - 0{\cdot}5(\lambda_0^2/\nu_0) \quad \text{for} \quad \nu_0 > 0{\cdot}1 \end{aligned}\right\} \quad . \quad (10)$$

$$\left.\begin{aligned} \bar{\Delta} &= \bar{v}/(\Omega R) \\ &= \lambda_0 - K \\ &= C_T'/2\nu_0 \text{ for } \nu_0 \text{ large (Momentum theory)} \end{aligned}\right\} \quad (11)$$

$$\simeq C_T'(1{\cdot}2 - \nu_0^2)/2\nu_0 \text{ (}see\text{ equation (}d\text{) of Appendix 1)}$$

$$\left.\begin{aligned} i &= \tan^{-1}(K/\mu) \\ &= \chi + \tau_c \end{aligned}\right\} \quad . \quad . \quad (12)$$

In (11) the approximation is chosen to fit experimentally determined values for hovering and to approach Momentum Theory for large ν_0.

1.2. SOLUTION OF THE PERFORMANCE EQUATION

A solution is sought giving the forward flight performance of the helicopter in terms of known or assumed parameters.

The most flexible method is by use of the "Energy-Account," first used by Glauert[2]. The main object of introducing this refined method is to take "account" of large vertical drag forces which arise in cases where stub wings are fitted.

Following the procedure in Ref. 1 we write down the balance of energy:

$$P = WV_c + T'\bar{v} + 0{\cdot}125\rho bc\Omega^3R^4\sqrt{(1 + 4{\cdot}5\nu_0^2)} + \\ + D_{PV}\sin(\psi + \chi) + D_{PH}\cos(\psi + \chi) \quad . \quad (13)$$

where

P power required at rotor shaft . (13*a*)

WV_c climb energy of helicopter . (13*b*)

$T'\bar{v}$ energy expended in accelerating induced velocity . (13*c*)

$0{\cdot}125\rho bc\Omega^3R^4\delta$ energy used in rotating rotor due to profile drag . (13*d*)

$0{\cdot}125 \times 4{\cdot}5\nu_0^2\rho bc\Omega^3R^4$ energy used in translating rotor due to profile drag . (13*e*)

D_{PV}' energy expended in overcoming parasitic drag. . . . (13*f*)

In the above it should be noted that:

(*i*) Expression (13*d*) is due to Squire[1].

(*ii*) Expression (13*e*) is due to Squire also, with an additional allowance due to Glauert[2] to account for radial flow effects.

(*iii*) The value of δ is estimated by methods described in Section 3 of this report for C_{D_0}' in Sheet D of the blade loading calculations.

Experience has shown that δ is given by

$$\delta = C_{D_0}' \quad . \quad . \quad . \quad . \quad . \quad (13g)$$

where C_{D_0}' is the value of C_{D_0}' at $x = 0{\cdot}68R$ and 0° azimuth; and is appropriate to an average lift coefficient $\bar{C}_L$ given by

$$\bar{C}_L = (nW/b)/[\rho S(\Omega R)^2 \times 0{\cdot}68^2] \quad . \quad (13h)$$

S plan area of blade. . . . (13*j*)

(*iv*) Practical application of the strip theory methods described in Section 2 indicates that the factor 4·5 in expression (13*e*) should be raised to 6·0 approximately.

Using expressions (5) we rewrite (13): —

$$P = WV_c + T'\bar{v} + 0{\cdot}125bc\rho\Omega^3R^4\delta(1 + 4{\cdot}5\nu_0^2) + \\ + D_{PV}V'(\lambda_0/\nu_0) + D_{PH}V'(\mu/\nu_0). \quad . \quad (14)$$

This may be rewritten in non-dimensional form using the definitions in the notation:

$$C_Q = C_W\frac{V_c}{\Omega R} + C_T'\bar{\Delta} + 0{\cdot}125\sigma\delta(1 + 4{\cdot}5\nu_0^2) + \\ + 0{\cdot}5C_{DPV}(\lambda_0^3/\nu_0)(\mu/\cos i) + 0{\cdot}5C_{DPH}(\mu^3/\nu_0)(\mu/\cos i)$$

or

$$C_Q = C_W\frac{V_c}{\Omega R} + C_T'\bar{\Delta} + 0{\cdot}125\sigma\delta(1 + 4{\cdot}5\nu_0^2) + \\ + 0{\cdot}5C_{DPV}\lambda_0^3 + 0{\cdot}5C_{DPH}\nu_0^3 + 0{\cdot}25C_{DPH}(\lambda_0^4/\nu_0) - \\ - 0{\cdot}75C_{DPH}\lambda_0^2\nu_0, \quad . \quad . \quad (15)$$

after substitution from (10).

In level flight, $V_c = 0$.

Hence

$$C_Q = C_T'\bar{\Delta} + 0{\cdot}125\sigma\delta(1 + 4{\cdot}5\nu_0^2) + 0{\cdot}5C_{DPV}\lambda_0^3 + \\ + 0{\cdot}5C_{DPH}\nu_0^3 + 0{\cdot}25C_{DPH}(\lambda_0^4/\nu_0) - 0{\cdot}75C_{DPH}\lambda_0^2V_0. \quad (16)$$

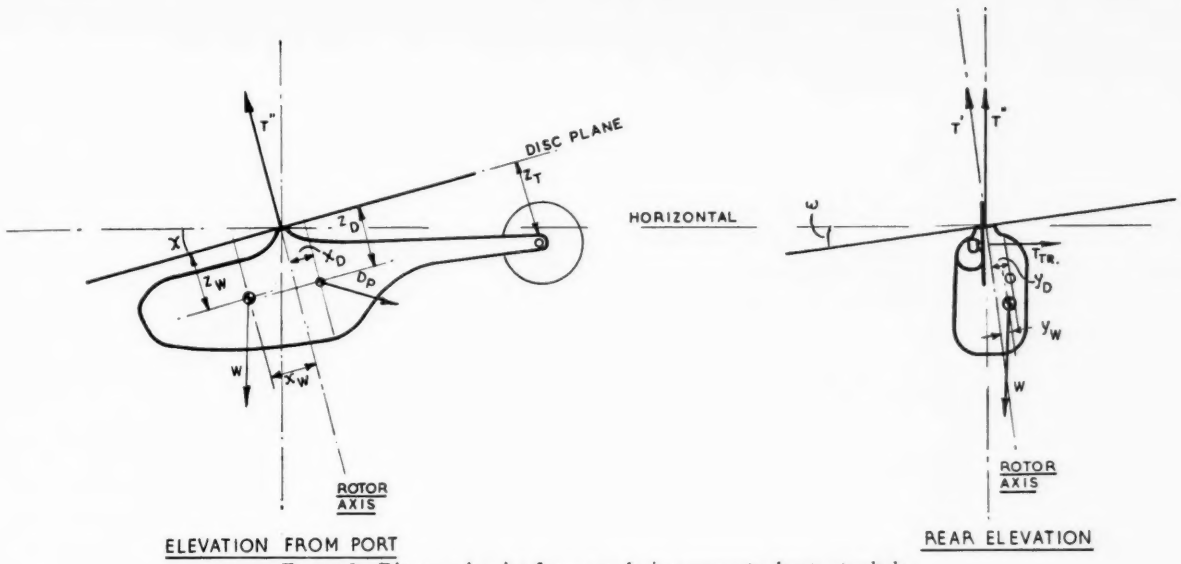


FIGURE 2. Diagram showing forces producing moments about rotor hub.

Using the substitutions given in Appendix I for $C_T'^2$ and $\bar{\Delta}$ we get: —

$$C_Q=\{(1\cdot2-\nu_0^2)C_W^2/2\nu_0\}+\{(1\cdot2-\nu_0^2)C_{DPV}\lambda_0^2/2\nu_0\}+ +0\cdot125\sigma\delta(1+4\cdot5\nu_0^2)+0\cdot5C_{DPV}\lambda_0^3+ +0\cdot5C_{DPH}\nu_0^3+0\cdot25C_{DPH}(\lambda_0^4/\nu_0)-0\cdot75C_{DPH}\lambda_0^2\nu_0$$

or $$A_1\lambda_0^4+B_1\lambda_0^3+C_1\lambda_0^2+D_1\lambda_0+E_1=0 \quad . \quad (17)$$

where

$$\left.\begin{aligned} A_1&=0\cdot25C_{DPH}/\nu_0\\ B_1&=0\cdot5C_{DPV}\\ C_1&=\{(1\cdot2-\nu_0^2)C_{DPV}/(2\nu_0)\}-0\cdot75C_{DPH}\nu_0\\ D_1&=0\\ E_1&=\{(1\cdot2-\nu_0^2)C_W^2/(2\nu_0)\}+0\cdot125\sigma\delta(1+4\cdot5\nu_0^2)+ +0\cdot5C_{DPH}\nu_0^3-C_Q. \end{aligned}\right\} \quad . \quad . \quad . \quad (17a)$$

1.3. SECOND APPROXIMATION TO ROTOR THRUST

Figure 2 shows an elevation from starboard and one from the rear of the helicopter. When the nature of the main rotor drive requires a torque reaction to balance, then the force T_{TR} which this generates is an additional debit to main rotor thrust C_T'.

For performance estimation it is sufficient to write $C_T'=C_T''$ or $\cos\omega=1\cdot0$ so that equations (6), (7), (8), and so on, follow.

$$\omega=\cos^{-1}(C_T''/C_T') \quad . \quad . \quad . \quad . \quad . \quad (27)^*$$

or $$\omega=\sin^{-1}(C_{T_{TR}}/C_T') \quad . \quad . \quad . \quad . \quad . \quad (28)$$

$$C_T'=C_T''/\cos\omega =[C_W\cos\chi+0\cdot5C_{DPV}\lambda_0^2+0\cdot5C_{DPH}\mu\lambda_0]/\cos\omega. \quad . \quad . \quad . \quad (29)$$

1.4. MOMENTS ABOUT ROTOR HUB

It is convenient to resolve moments about the rotor hub as it is easier to calculate the reacting airloading moments about this axis.

Pitching Moment: Contributions to pitching moment mainly stem from weight and parasitic drag. In some cases a C_{m_0} term due to fuselage might be significant.

Moment balance is obtained when

$$M_a+M=0 \quad . \quad . \quad (30)$$

where M moment due to air loading

M_a moment due to external forces.

From Fig. 2 we may write

$$M=W(x_w\cos\chi-z_w\sin\chi)+D_P(z_D\mu/\nu_0)-x_D(\lambda/\nu_0)$$

so that non-dimensionally:

$$C_m=\frac{C_W}{R}(x_w\cos\chi-z_w\sin\chi)+\tfrac{1}{2}C_{DP}\frac{\nu_0}{R}(\mu z_D-\lambda_0 x_D) \quad (31)$$

in general $\lambda_0 x_D$ is negligible and we write

$$C_m\simeq\frac{C_W}{R}(x_w\cos\chi-z_w\sin\chi)+\tfrac{1}{2}C_{DP}\frac{\nu_0\mu}{R}z_D. \quad . \quad (31a)$$

This equation (31*a*) implies an infinitely stiff blade. In practice blade deflections will take place due to the lift loadings, and these may be thought of as being made up of a steady loading plus a loading variable with azimuth. This latter contribution causes the rotor blade to deflect differentially relative to the coning angle and thus produce variable moments about the hub due to centrifugal forces.

Knowing the characteristics of the blade structure it is possible to formulate a relationship between an arbitrary trapezoidal loading distribution and hub bending moment. This in turn is converted into a pitching moment coefficient by a single factor. This pitching moment coefficient applies to a blade of zero inertia, and with zero aerodynamic damping due to flapping. A separate investigation into the effective rigidity of the rotor system will give the magnification

*Equations (18) to (26) are omitted from this paper, the subject matter to which they referred being irrelevant.

of these moments, including any excitation effects due to the natural frequency of the blade being near the forcing frequency. Note that such an investigation will show the control advance necessary to offset these effects.

We may write therefore

$$C_m \simeq \frac{C_W}{R}(x_w \cos\chi - z_w \sin\chi) + 0{\cdot}5C_{DP}\ \frac{v_0}{R}\mu z_D + C_{mC.F.} \quad . \quad . \quad (31b)$$

where $C_{mC.F.}$ is the pitching moment coefficient due to the centrifugal forces arising from the variable component of the thrust loading.

Rolling Moment: There is a third contribution to rolling moment due to torque reaction, as shown in Fig. 2.

$$l = T_{TR}z_t - Wy_w \cos\omega - D_{PV}y_D \cos\omega - Wz_w \sin\omega$$

i.e.

$$C_l = \frac{C_{TTR}}{R}z_t - \frac{C_W}{R}y_w \cos\omega - \frac{C_W}{R}z_w \sin\omega - \\ - \tfrac{1}{2}C_{DPV}\frac{v_0^2}{R}y_D \cos\omega. \quad . \quad . \quad . \quad (32)$$

In a similar manner the rolling moment balance equation is modified to:

$$C_l = \frac{C_{TTR}}{R}z_t - \frac{C_W}{R}(y_w \cos\omega + z_w \sin\omega) - \\ - 0{\cdot}5\frac{C_{DPV}}{R}v_0^2 y_D \cos\omega + C_{lC.F.}. \quad . \quad . \quad (32a)$$

It should be noted that the convention of signs used above is governed by that used for deriving C_{ma} and C_{la}, the pitching and rolling moments due to air loading.

Where azimuth angle is measured from aft with starboard blade advancing then:

Nose down moments are positive,
Rolling moments to port are positive.

Section 2

2.1. In this Section the components of the incident flow, tangential and normal to the surface of rotor disc, are derived in terms of azimuth angle, coning angle, and radial position on the blade.

By vectorial addition of the tangential component of incident velocity and the rotational velocity, a resultant velocity is deduced acting at an angle of "sweep."

Work proceeds from this point by considering the blade divided radially into elementary strips, each strip being acted upon by an incident flow at the angle of sweep mentioned above. In the general case there is an analogy between the rotor blade in forward flight and the "crescent" wing, in as much as the sweep is least at the tip and a maximum at the root of the blade.

The incidence of the element (*i.e.* its angle of attack) is deduced by taking into account the blade pitch, the component of incident flow normal to the coned disc and the induced velocity.

The induced velocity variation over the disc can be quite arbitrary, but for this work it has been decided that the expression derived by Drees[7] is satisfactory for moderate values of the tip-speed ratio.

2.2. The elemental lift and drag equations are developed taking into account the effects of sweep as given in Refs. 8 and 9.

The assumption that the flow is two-dimensional is approximately true for root and intermediate elements, but a correction factor is required for elements near the tip. Use is made of the analysis by Lock and Teatman[10] of the modified vortex theory of Goldstein, and Prandtl's approximation to it. Consideration of the mechanism by which this tip effect operates shows that, although the concept of integrating the differential lift-forces out to a point inboard of the tip is generally admissible, serious errors are involved by putting the same integration limit on induced drag. An extension to Lock's work is made therefore to deal with this aspect.

The elemental drag forces act along the line of the resultant velocity (*i.e.* at an angle of sweep), and expressions are developed for the elemental torque and H forces. Similar treatment of the elemental lift equation gives the thrust and pitching and rolling moments about axes through the hub.

2.3. The main formulae derived are collected and re-presented.

2.1. RESOLUTION OF LOCAL VELOCITIES AT ROTOR DISC

Figure 3(*a*) repeats the picture given in Fig. 1 of the disc set at an arbitrary angle to the incident velocity V.

The blades, three of which are indicated at azimuth angles of 0°, ϕ° and 180° are set at an arbitrary coning angle β, which may or may not be constant with azimuth angle. To point this up, the tip-path plane is shown not parallel to the disc plane.

The velocity, angles and forces in Figs. 3(*a*)-3(*c*) are positive as drawn but the argument now developed is valid for negative values, except negative values of V_r.

Consider the components of the incident flow tangential to and normal to the surface of the cone described by the blades.

Then by geometry we have: —

$$V_t' = V\cos(i + \beta\cos\phi) \quad . \quad . \quad (33)$$

$$V_n = V\sin(i + \beta\cos\phi). \quad . \quad . \quad (34)$$

Consider now the element δr of the blade at azimuth ϕ, and let δr be at distance r from the hub, as shown in Figs. 3(*a*) and 3(*b*).

There are two tangential velocities at the element, the component of forward speed V_t' and the rotational velocity Ωr, which add vectorially to give a resultant velocity V_r where by geometry:

$$V_r = [(\Omega r + V_t'\sin\phi)^2 + (V_t'\cos\phi)^2]^{0{\cdot}5}. \quad . \quad (35)$$

We see that V_r acts at a sweep angle γ relative to the direction normal to the blade leading edge, and by reference to Fig. 3(*b*)

$$\gamma = \sin^{-1}(V_t'\cos\phi/V_r). \quad . \quad . \quad (36)$$

Inspection of equations (35) and (36) shows that elements at the tip of the blade are effectively at a smaller angle of sweep than those nearer the hub. It is possible therefore to say that in general a blade in forward flight experiences flow conditions analogous to those in a "crescent" fixed wing. (A description of two types of crescent wing was given in Ref. (6) and illustrated in Figs. 12 and 13 of that paper). The radial velocity gradients make this analogy of limited use and it should not be developed further.

However, in the following development the elements are treated as parts of an infinite sheared wing at sweep angle γ and this factor is taken into account in connection with incidence, lift curve slope, profile drag, and so on.

Equation (36) may be expressed in terms of non-dimensional velocity coefficients:
By definition

$$\mu = V\cos i/(\Omega R) \quad . \quad . \quad . \quad . \quad . \quad (10)$$

$$K = V\sin i/(\Omega R). \quad . \quad . \quad . \quad . \quad . \quad (37)$$

Hence in (33) and (34)

$$u' = V_t'/(\Omega R) = \mu\cos(i+\beta\cos\phi)/\cos i \quad . \quad (38)$$

$$K' = V_n/(\Omega R) = \mu\sin(i+\beta\cos\phi)/\cos i. \quad . \quad . \quad (39)$$

In (35) writing $x = r/R$

$$\mu'' = V_r/(\Omega R) = [(x+\mu'\sin\phi)^2 + (\mu'\cos\phi)^2]^{0.5}. \quad (40)$$

In (36)

$$\gamma = \sin^{-1}(\mu'\cos\phi/\mu''). \quad . \quad . \quad (41)$$

It is worth noting here that μ'' is set up only for positive velocities. Special treatment would be required for a blade element in a region of reversed flow.

Similarly the expression for K' is not rigorous in as much as a factor $\leqslant \cos\beta$ should be applied, maxima occurring at $\phi = 90°$ and $270°$.

In addition to the normal components V_n, there is the induced velocity v, *see* Fig. 3(*c*). Strictly the component for v acting normal to the disc-cone surface should be used so that the total velocity is

$$\lambda\Omega R = V_n + v\cos B$$

or in non-dimensional form and for $\beta \longrightarrow 0$

$$\lambda = K' + \Delta\cos B \simeq K' + \Delta. \quad . \quad . \quad (42)$$

Because the induced velocity is not uniform across the rotor disc it is necessary to assume some variation relative to the momentum theory value which is given by

$$\overline{\Delta} = \overline{v}/(\Omega R) = C_T/(2\nu_0). \quad . \quad . \quad (43)$$

The treatment of the problem by Drees[7] is well suited for inclusion in this method and he shows that for values of μ not near zero the following holds:

$$\Delta = v/(\Omega R) = \overline{\Delta}\{(1 + x(1{\cdot}15\cos\phi - 2\mu\sin\phi)\}. \quad (44)$$

It should be noted that the wind tunnel tests of Ref. (16) indicate an asymmetric downwash distribution of a type similar to that due to the $-2\mu\sin\phi$ term in (44).

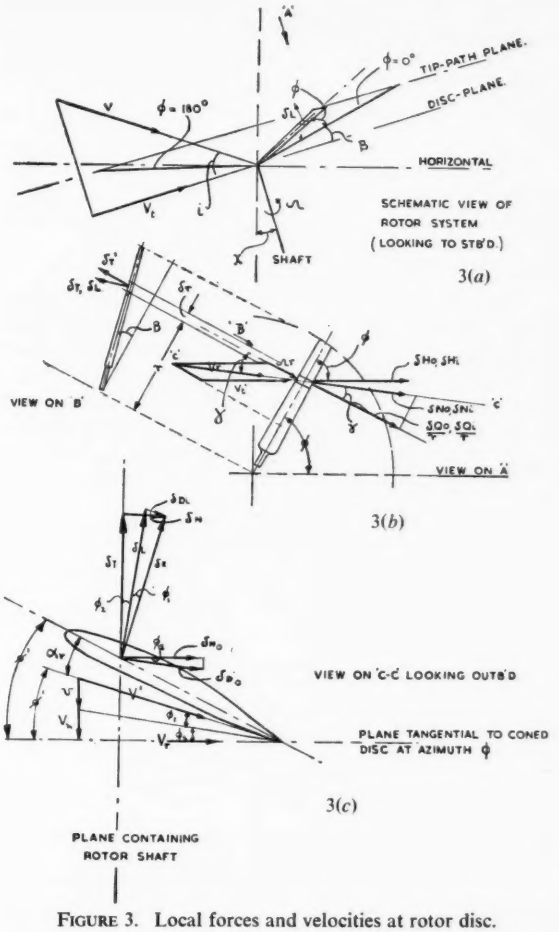


FIGURE 3. Local forces and velocities at rotor disc.

Referring now to Fig. 3(*c*) it is seen that the tangential and normal velocities V_r and $(V_n + v)$ respectively give a resultant velocity V'

i.e. $$\nu = V'/(\Omega R) = [V_r^2 + (V_n+v)^2]^{0.5}/(\Omega R) = [\mu''^2 + \lambda^2]^{\frac{1}{2}}$$
. . . . . (45)

i.e. $$\nu = [\mu''^2 + (K'+\Delta)^2]^{0.5}$$

or $$\nu^2 = \mu''^2\left[1 + \left(\frac{K'+\Delta}{\mu''}\right)^2\right]. \quad . \quad . \quad . \quad . \quad (46)$$

It can be shown that for $x > 0{\cdot}25$ and for $V_c \simeq 0$ the factor

$$\left[1 + \left(\frac{K'+\Delta}{\mu''}\right)^2\right] \longrightarrow 1{\cdot}0$$

and therefore in the following treatment it is assumed that

$$\nu \simeq \mu''. \quad . \quad . \quad . \quad (47)$$

A justification for this lies in the fact that the factor is only appreciably different from unity when μ'' is so small that the forces involved are negligible (*see also* equation 65).

The angle of incidence of the blade element is shown in Fig. 3(*c*) as

$$\alpha_r = \theta' - \phi' \quad . \quad . \quad . \quad . \quad (48)$$

where $$\theta' \simeq \theta \cos\gamma \quad . \quad . \quad . \quad . \quad (49)$$

$$\phi' = \tan^{-1}(\lambda/\mu'') \simeq \lambda/\mu''. \quad . \quad . \quad (50)$$

Hence $$\alpha_r = \theta\cos\gamma - \lambda/\mu''$$

or $$\alpha_r = \theta\cos\gamma - (K' + \Delta)/\mu'' \quad . \quad (51)$$

2.2. The Forces on the Blade Elements

Lift: It was shown in the previous section that the blade element may be considered as swept with the angle of sweep a variable in azimuth and radial position.

Because the blade profile is swept, unlike a swept wing where the profiles are generally maintained correct in line of flight, *i.e.* unswept profiles, it is necessary to correct the lift curve slope. It is shown in Ref. (8) that $a_{0\gamma}$ the swept, and a_0 the unswept, two-dimensional lift-curve slopes are related by

$$a_{0\gamma} = a_0 \cos\gamma = C_L/\alpha_r. \quad . \quad . \quad (52)$$

It should be noted that as the two-dimensional lift-curve slope is not independent of Reynolds number in practice the correction given in Ref. (8) is applied.

Writing now the incremental lift equation

$$\delta L = C_L \tfrac{1}{2}\rho c V'^2 \delta r$$
$$= a_0 \cos\gamma \alpha_r \tfrac{1}{2}\rho c V'^2 R\delta x$$
$$= \frac{a_0}{2}\cos\gamma c(\theta' - \phi')\rho\nu^2(\Omega R)^2 R\delta x$$

or $$\delta L = \frac{a_0}{2b}\sigma_x \cos\gamma(\theta' - \phi')\pi R^2\rho(\Omega R)^2\nu^2\delta x \quad . \quad (53)$$

when substitutions in the right hand side have been made from (52), (48), (45) and

$$\sigma_x = bc/(\pi R) \quad . \quad . \quad . \quad (54)$$

with the approximation of (47) we have

$$\delta L \simeq \frac{a_0}{2b}\sigma_x \cos\gamma(\theta' - \phi')\pi R^2\rho(\Omega R)^2\mu''^2\delta x. \quad . \quad (55)$$

This force acts normal to incident flow less induced effects and hence is a component of the result force δR such that

$$\delta L = \delta R \cos\phi_1. \quad . \quad . \quad . \quad (56)$$

This is shown in Fig. 3(*c*), ϕ_1 being the induced part of ϕ'. The induced drag component δD_i is given by

$$\delta D_i = \delta L \tan\phi_1 \simeq \delta L\phi_1 \quad \text{as} \quad \phi_1 \quad \text{is small.} \quad (57)$$

Equations (53) and (57) assume two-dimensional flow from $x=0$ to $x=1$. This condition obtains only with an infinite number of blades.

A working method for estimating the effect on lift and thrust of a finite number of blades is given by Lock and Yeatman[16]. This is based upon an approximation by Prandtl to the modified vortex theory of Goldstein. This method was investigated[15] with a view to application to the results given by the use of this report.

Appendix III summarises the approach to the problem and shows that there are serious errors in putting the limit to the integration of induced torque at the same station as that which is deduced for thrust. Ref. 15 shows that in fact it is not possible to use a limit that will give a satisfactory approximation to tip-losses for forward flight cases.

Following Ref. 15 tip-losses are accounted for by the use of two factors (*see* Appendix III).

F Tip-loss factor in lift or thrust

G Tip-loss factor in induced drag or torque

giving:

$$\delta L = \frac{Fa_0}{2b}\sigma_x \cos\gamma(\theta' - \phi')\pi R^2\rho(\Omega R)^2\mu''^2\delta x \quad . \quad (55a)$$

$$\delta D_i = G\delta L\phi_1. \quad . \quad . \quad . \quad . \quad . \quad (57a)$$

Profile Drag:

There is a modification to the profile drag when a wing is swept, caused by the changes in the boundary layer conditions. The correction factors have been derived by Weber[9] who shows that for Reynolds number in the region of 10^6 and for normal transition

$$C_{D0\gamma}' = C_{D0}'\cos^{0.5}\gamma \quad . \quad . \quad . \quad . \quad (58)$$

the incremental drag equation is

$$\delta D_0' = C_{D0}'\cos^{0.5}\gamma\,\tfrac{1}{2}\rho c V'^2\delta r.$$

Therefore

$$\delta D_0' = \frac{C_{D0}'}{2b}\sigma_x\cos^{0.5}\gamma\pi R^2\rho(\Omega R)^2\mu''^2\delta x. \quad . \quad (59)$$

Thrust:

As shown in Fig. 3(*c*) the thrust force δT is normal to the tangent plane. The useful component of this is $\delta T'$, *see* Fig. 3(*b*), so that

$$\delta T' = \delta T\cos\beta$$
$$= [(\delta L - \delta D_i\tan\phi_2)\cos\phi_2 - \delta D_0\sin\phi_2]\cos\beta.$$

From (57) and assuming:

$$\cos\beta = 1, \qquad \delta D_0\sin\phi_2 = 0$$

we have

$$\delta T' \simeq \delta L(1 - \tan\phi_1\tan\phi_2)\cos\phi_2. \quad . \quad (60)$$

From Fig. 3(*c*) and by geometry

$$\tan\phi_1 = \Delta/[\{\lambda(\lambda - \Delta)/\mu''\} + \mu''] \simeq \Delta/\mu'' \quad (61)$$

$$\tan\phi_2 = K'/\mu'' \quad . \quad . \quad . \quad . \quad (62)$$

Therefore

$$\cos\phi_2 = \mu''/(K'^2 + \mu''^2)^{\frac{1}{2}} \simeq \mu''/\nu \quad . \quad . \quad (63)$$

$$\delta T' \simeq \delta L\left\{1 - \frac{\Delta K'}{\mu''^2}\right\}\frac{\mu''}{\nu}. \quad . \quad . \quad (64)$$

Except where μ''^2 is small this factor is negligible and is in any case working in opposition to the factor given in (46).

Therefore $$\delta T' \simeq \delta L. \quad . \quad . \quad . \quad (65)$$

Hence from (55*a*)

$$\delta T' = F\frac{a_0\sigma_x}{2b}\cos\gamma(\theta' - \phi')\pi R^2\rho(\Omega R)^2\mu''^2\delta x$$

or in non-dimensional form

$$\delta C_T' = F \frac{a_0 \sigma_x}{2b} \cos\gamma\,(\theta' - \phi')\,\mu''^2 \delta x. \qquad (66)$$

Torque:

The induced and profile torques are considered separately.

Induced Torque:

The component of δR complementary to δT is δN_i, which as shown in Figs. 3(*b*) and 3(*c*) acts in the local air flow direction in plan view (*i.e.* at sweep angle γ) and at right angles to δT_0. Expressing δN_i in terms of δL, δD_i we have from (57*a*)

$$\delta N_i = G\delta T \tan\phi' \quad . \quad . \quad . \quad (67)$$

so that on resolving through angle γ we obtain the torque force $\delta Q_i/r$, giving

$$\delta Q_i = Gr\delta T \tan\phi' \cos\gamma,$$

so that upon substituting from (50) and (65) we have

$$\delta Q_i = Gr\delta L \left\{\frac{\lambda}{\mu''}\right\} \cos\gamma. \quad . \quad . \quad (68)$$

Hence in non-dimensional form with δL from (55*a*)

$$\left.\begin{aligned} &\delta C_{Qi} = Gx \frac{a_0}{2b} \sigma_x \cos^2\gamma\,(\theta' - \phi')\,\mu''^2 \left(\frac{\lambda}{\mu''}\right) \delta x \\ \text{or} \quad &\delta C_{Qi} = Gx\delta C_T' \left\{\frac{\Delta + K'}{\mu''}\right\} \cos\gamma \delta x. \end{aligned}\right\} \qquad (69)$$

Profile Torque:

In a similar manner we write

$$\delta N_0 = \delta D_0' \cos\phi_2 \quad . \quad . \quad . \quad (70)$$

$$\delta Q_0 = r\delta D_0' \cos\phi_2 \cos\gamma \quad . \quad . \quad (71)$$

so that in substituting (59) and (63) and making non-dimensional

$$\delta C_{Q0} = x \frac{C_{D0}'}{2b} \sigma_x \cos^{0.5}\gamma \mu''^2 (\mu''/\nu) \cos\gamma \delta x$$

so that with the approximation of (47): —

$$\delta C_{Q0} = x \frac{C_{D0}'}{2b} \sigma_x \cos^{0.5}\gamma \cos\gamma \mu''^2 \delta x. \quad . \quad . \quad (72)$$

Disc Drag Force:

This force, commonly called the H force, lies in the disc plane and is the component of $N_0 + N_i$ in line of flight. Hence by geometry (*see* Fig. 3(*b*))

$$\delta H_i = \frac{\delta Q_i}{r} \frac{\sin(\gamma+\phi)}{\cos\gamma} \quad . \quad . \quad . \quad (73)$$

$$\delta H_0 = \frac{\delta Q_0}{r} \frac{\sin(\gamma+\phi)}{\cos\gamma} \quad . \quad . \quad (74)$$

so that in non-dimensional terms: —

$$\delta C_{Hi} = \frac{\delta C_{Qi}}{x} \frac{\sin(\gamma+\phi)}{\cos\gamma} \quad . \quad . \quad (75)$$

$$\delta C_{H0} = \frac{\delta C_{Q0}}{x} \frac{\sin(\gamma+\phi)}{\cos\gamma} \quad . \quad . \quad (76)$$

Hub Moments due to Air Loadings:

These arise solely due to the thrust moments and are defined as

$$\delta M_a = \delta T r \cos\phi \quad . \quad . \quad . \quad (77)$$

$$\delta l_a = \delta T r \sin\phi \quad . \quad . \quad . \quad (78)$$

giving the non-dimensional expressions

$$\delta C_{ma} = \delta C_T' x \cos\phi \quad . \quad . \quad . \quad (79)$$

$$\delta C_{la} = \delta C_T' x \sin\phi. \quad . \quad . \quad . \quad (80)$$

2.3. SUMMARY OF FORMULAE

Force Equations:

If the incremental expressions for thrust torque, and so on, given in Section 2 are taken to the limit we get the following: —

From (66) $$dC_T'/dx = \frac{a_0}{2b} \sigma_x \cos\gamma \alpha_T \mu''^2 F. \quad . \quad (81)$$

From (69) $$dC_{Qi}/dx = (dC_T'/dx) \left(\frac{\Delta + K'}{\mu''}\right) \cos\gamma x G. \quad . \quad . \quad . \quad (82)$$

From (72) $$dC_{Q0}/dx = \frac{C_{D0}'}{2b} \sigma_x \cos^{0.5}\gamma \cos\gamma \mu''^2 x. \quad (83)$$

From (75) $$dC_{Hi}/dx = (dC_{Qi}/dx) \sin(\gamma+\phi)/(x\cos\gamma). \quad . \quad . \quad . \quad (84)$$

From (76) $$dC_{H0}/dx = (dC_{Q0}/dx) \sin(\gamma+\phi)/(x\cos\gamma). \quad . \quad . \quad . \quad (85)$$

From (79) $$dC_{ma}/dx = (dC_T'/dx)\, x \cos\phi. \quad . \quad (86)$$

From (80) $$dC_{la}/dx = (dC_T'/dx)\, x \sin\phi. \quad . \quad . \quad (87)$$

Velocity:

Collecting and re-presenting the equations in Section 2: —

From (16) $$\mu = V\cos i/(\Omega R). \quad . \quad . \quad . \quad (88)$$

From (37) $$K = V\sin i/(\Omega R). \quad . \quad . \quad . \quad . \quad (89)$$

From (38) $$\mu' = V_t'/(\Omega R) = \mu\cos(i + \beta\cos\phi)/\cos i. \quad . \quad . \quad . \quad (90)$$

From (39) $$K' = V_n'/(\Omega R) = \mu\sin(i + \beta\cos\phi)/\cos i. \quad . \quad . \quad . \quad (91)$$

From (40) $$\mu'' = V_r/(\Omega R) = [(x + \mu'\sin\phi)^2 + (\mu'\cos\phi)^2]^{0.5} \quad . \quad . \quad . \quad (92)$$

From (42) $$\lambda = K' + \Delta\cos\beta \simeq K' + \Delta. \quad . \quad . \quad (93)$$

From (43) $$\bar{\Delta} = \bar{v}/(\Omega R) = C_T/(2v_0). \quad . \quad . \quad (94)$$

From (44) $$\Delta = v/(\Omega R) = \bar{\Delta}[1 + x(1{\cdot}15\cos\phi - 2\mu\sin\phi)]. \quad . \quad . \quad . \quad (95)$$

From (45)-(47)

$$v = V'/(\Omega R) = \mu''\left[1 + \left(\frac{K' + \Delta}{\mu''}\right)^2\right]^{0.5} \simeq \mu''. \quad . \quad . \quad . \quad (96)$$

Angles:

Collecting and re-presenting the equations in Section 2: —

From (41) $$\gamma = \sin^{-1}(\mu'\cos\phi/\mu''). \quad . \quad . \quad (97)$$

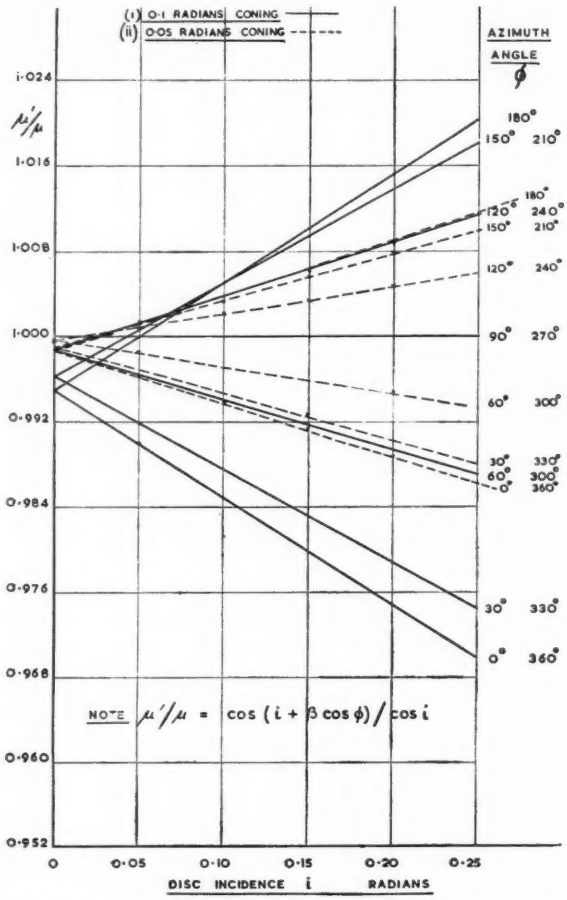


FIGURE 4. Relationship between velocity ratio μ'/μ and disc plane incidence.

From (49) $\theta' \simeq \theta \cos\gamma$. . . . . (98)

From (50) $\phi' = \tan^{-1}(\lambda/\mu'') \simeq \lambda/\mu'' \simeq (K' + \Delta)/\mu''$. . . . . (99)

From (51) $\alpha_r = \theta' - \phi' = \theta \cos\gamma + \{(K' + \Delta)/\mu''\}$. (100)

Section 3

The procedure which has been evolved is described with the aid of sample working-sheets which are appended. It is realised that improvements in the numerical analysis could be made and this description and these sheets are mainly for the purpose of illustration.

3.1. PROCEDURE

The expressions evolved in Section 2 may be used for a variety of purposes connected with blade stressing. For example the loadings in steady flight may be evaluated. Accelerated flight cases may be treated similarly, the main changes being in the induced velocity field and in the cyclic pitch variations to balance. It would be possible also to include the curvature of the flight path for these "g" cases.

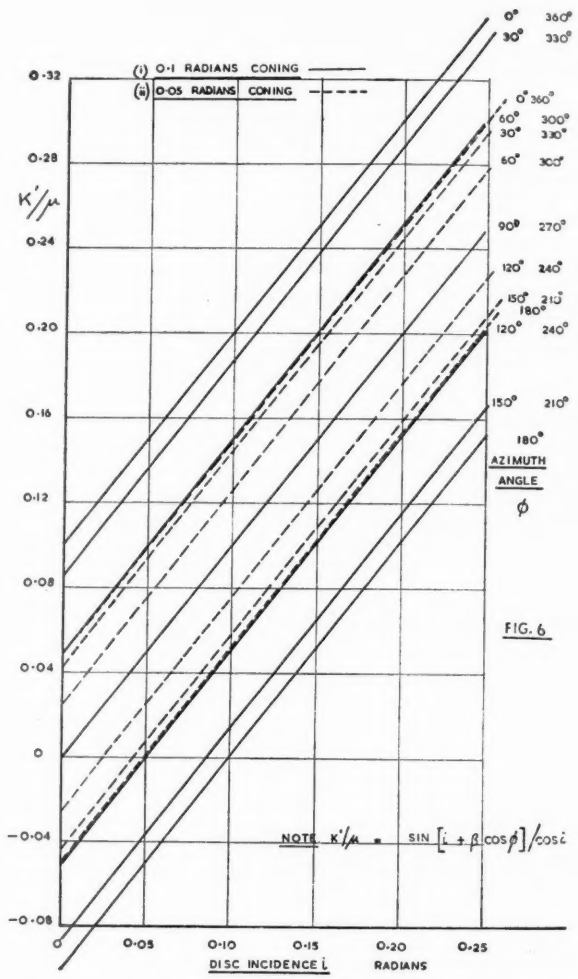


FIG. 6

FIGURE 5. Relationship between velocity ratio K'/μ and disc incidence.

The work carried out for these cases gives information whereby the following may be readily obtained:

(*a*) loadings due to gusts,

(*b*) loadings due to collective pitch manoeuvres,

(*c*) loadings due to cyclic-pitch manoeuvres.

In this section the procedure which has been evolved for determining the basic information is described and illustrated by reference to the working sheets which are appended. These sheets are set out for use with a desk calculator but are thought to be capable of improvement with experience.

The loadings are derived for blades at arbitrary azimuth angles. In this work the information that can be presented graphically in a general manner has been given in Figs. 4-9 for azimuth angles of 0° to 360° by 30° steps. This is most convenient for evaluating the combined loadings of 2, 3, 4 or 6-bladed rotors.

CALCULATION OF BLADE LOADINGS — FORWARD FLIGHT.

AZIMUTH ANGLE ϕ = COS ϕ = (1)
SIN ϕ = (2)

FROM WORK REF.
ΩR = FT./SEC; μ = K = λ_0 = ν_0 = $\bar{\Delta}$ = (3)

i = RADS. χ = RADS. τ_c = RADS.

1·15 COS ϕ = 2μ SIN ϕ =
1·15 COS ϕ − 2μ SIN ϕ = $[(\Delta/\bar{\Delta}) - 1]/\lambda$ (4)
FROM D.A. 3158 μ'/μ = μ' = (5)
FROM D.A. 3159 K'/μ = K' = (6)

x	0·2	0·3	0·4	0·5	0·6	0·7	0·8	0·9	1·0	
CHORD										(7)
$6\pi b = \frac{c}{\pi R}$										(8)
$\Delta\theta$ DUE TO TWIST										(9)
μ'/μ' (D.A. 3160-1)										(10)
μ'' FROM (5) & (10)										(11)
μ''^2										(12)
x/μ'										(13)
γ (D.A. 3162-3)										(14)
SIN γ										(15)
COS γ										(16)
COS $(\gamma-\phi)$ = (1)(16) + (2)(15)										(17)
SIN $(\gamma+\phi)$ = (1)(15) + (2)(16)										(18)
$\Delta\theta$ COS γ										(19)
$\Delta/\bar{\Delta}$ FROM (4)										(20)
Δ										(21)
$\Delta + K'$										(22)
$(\Delta + K')/\mu''$										(23)
(19) − (23)										(24)
θ_0 COS γ FOR θ_0 = ·05										(25)
α_γ = (24) + (25)										(26)
θ_0 COS γ FOR θ_0 = ·10										(27)
α_γ = (24) + (27)										(28)
θ_0 COS γ FOR θ_0 = ·20										(29)
α_γ = (24) + (29)										(30)
θ_0 COS γ FOR θ_0 = ·30										(31)
α_γ = (24) + (31)										(32)
F (D.A. 3209)										(33) *
*FAT χ = 0·984										(34)
COS^2 γ										(35)
$\sigma_\chi \mu''^2$ = (8)(12)										(36)
χ SIN ϕ										(37)
χ COS ϕ										(38)
a_0 COS γ = (42) (16) = $a_0\phi$										(39)
COS$^{3/2}$ γ										(40)
Δa_0 at R.N. (D.A.										(41)
a_0 = $(a_0)_\xi$ − Δa_0 (SHEET B)										(42)
R.N. FROM SHEET D										(W)

SHEET A.

3.2. SHEET A

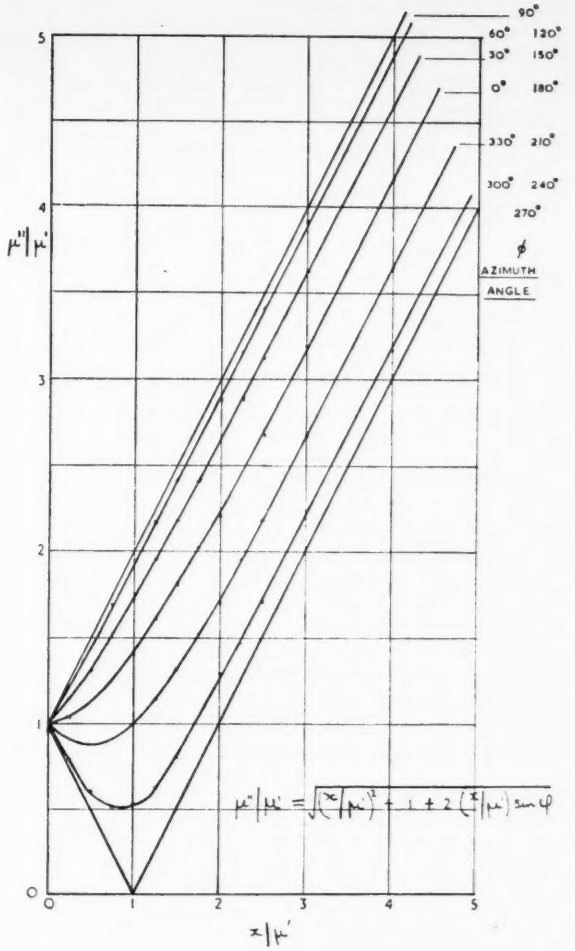
Each azimuth angle for which loading information is required has a sheet A prepared for it. The performance estimation method derived in Section 1 is used to obtain the components of incident velocity which are expressed in non-dimensional coefficients μ, K, λ_0, ν_0. This solution gives the required angles i, χ and τ_c.

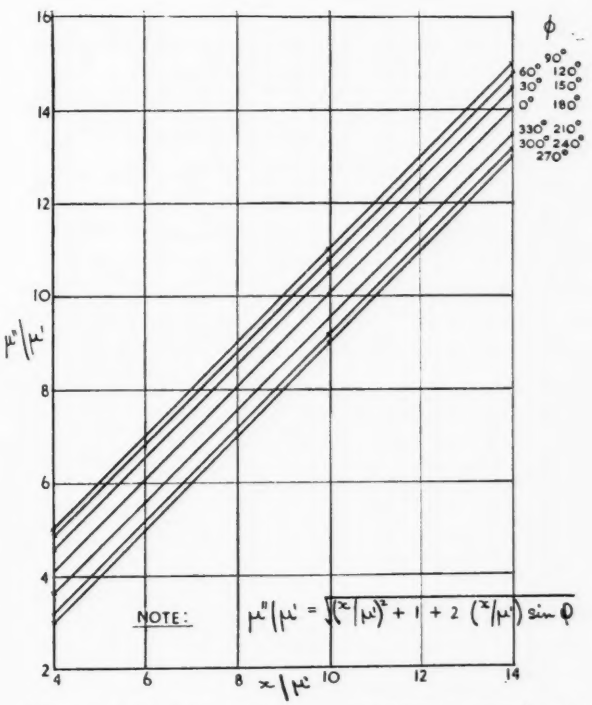
From (94) we obtain $\bar{\Delta}$ and (95) gives $\Delta/\bar{\Delta}$ in terms of x.

The span of the blade is divided into nine equal elements, starting at $x = 0{\cdot}2R$. This number is the least which can be used if accuracy is desired, and the arbitrary limits of $0{\cdot}2$ and $1{\cdot}0R$ facilitate the numerical integrations using a desk type calculating machine.

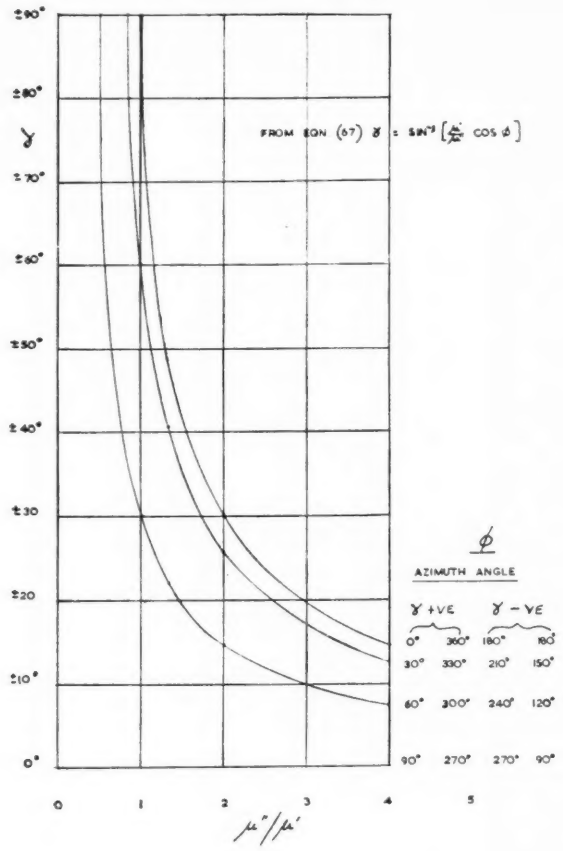
From (90) we may derive μ'/μ and hence μ'. The general derivation is given in Appendix II, Table I and Fig. 4 is prepared from it. Similarly K'/μ is obtained from (91) and hence K'. The general derivation is given in Table II of Appendix II and plotted in Fig. 5.

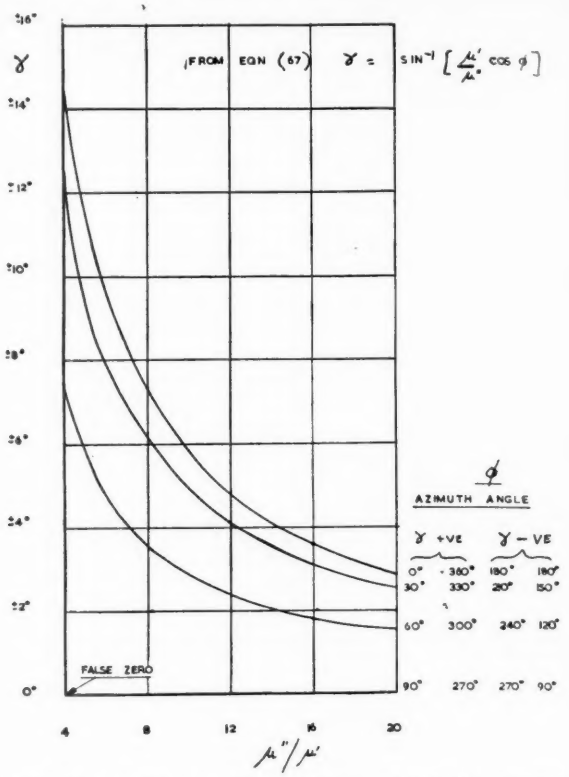
In a similar manner μ''/μ' is obtained from (92),





FIGURES 6 (*above*) and 7 (*below*). Relationship between μ''/μ' and x/μ' for several azimuth angles ϕ.





FIGURES 8 (*above left*) and 9 (*above*). Relationship between effective sweep γ and velocity ratio μ''/μ' for several blade azimuth angles.

giving μ''. The general derivation is given in Table III of Appendix II and plotted on Figs. 6 and 7.

The sweep angle γ is given by (97) and is derived in general form in Table II of Appendix II and plotted in Figs. 8 and 9.

Lines 15-19 give some geometric values required at a later stage, 23-32 is a solution of α_T from (100) in terms of arbitrary θ_0. θ_0 is a datum pitch angle and $\Delta\theta$, line 9, is the twist of the blade relative to this datum.

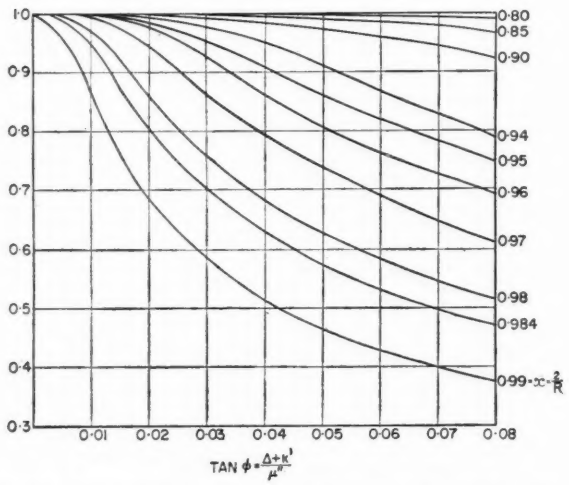


FIGURE 10. Thrust tip loss factor F plotted against induced angle and spanwise position for three-bladed rotor or airscrew.

Further useful quantities follow on lines 33 to 40. Line 33 is used for the evaluation of the tip loss factor F using Fig. 10 from Ref. 15. At the tip $x=1{\cdot}0$ the value of F for $x=0{\cdot}984$ is inserted. It was determined in Ref. 15 that a good approximation to the exact tip-effect is attained by this means.

The lift curve slope is evaluated by the method of Collingbourne[8]. This makes use of the theoretical lift-curve slope given by

$$a_0 = 2\pi + (4{\cdot}75 + 0{\cdot}02\varepsilon_T)t/c \quad . \quad . \quad (101)$$

which is corrected by a function Δa_0 which is dependent upon Reynolds number and trailing edge angle. The value of Δa_0 is read from Fig. 11 which is appended.

It is recommended that a further reduction of 5 per cent. be made to a_0 to take some account of the effect of the oscillatory wake induced by the blade motion and to account for a fall-off near $C_{L\,max}$.

3.3. SHEET B

Usually values of θ_0 only are evaluated. The required entries are made on lines 43, 54 and 65 from Sheet A. If α_T is around 0·25 radians it is worthwhile checking that C_L is not much above the expected steady

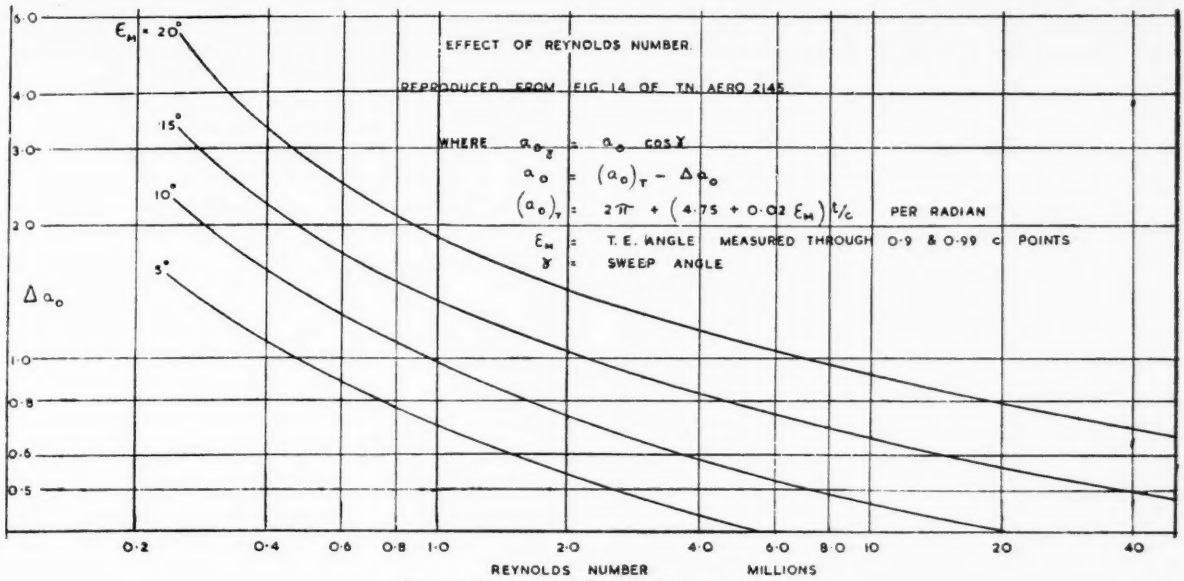


FIGURE 11. Calculation of lift-curve slope.

value of $C_{L\,max}$. As an arbitrary limit $C_L/C_{L\,max} = 1{\cdot}15$ is suggested, as some Wagner effect may be expected. The method of calculating $C_{L\,max}$ is given for Sheet D.

dC_T'/dx follows from equation (81), and C_T' by integration.

dC_{la}/dx follows from equation (87) and C_{la} by integration.

dC_{ma}/dx follows from equation (86) and C_{ma} by integration.

3.4. SHEET C

Results for one set of azimuth angles of a three-bladed rotor are collected on Sheet C from the respective sheets A and B.

If the method of Ref. (15) has not been used, *i.e.* factor $F = 1{\cdot}0$ throughout, then it is possible to approximate to the tip loss as given below.

Usually it is convenient to use the more exact method and in this case lines (83) to (88), (97) and (101) should be omitted.

Tip Losses:

In general there is a limit of integration which is assumed to account for the loss of lift at the tip where flow becomes three-dimensional in character. Sources such as Ref. 3 give this limit as

$$B = 1 - \frac{\sqrt{(2C_T')}}{b} \quad . \quad . \quad . \quad (102)$$

and the equations affected are those for dC_T'/dx, (81); dC_{la}/dx, (87); dC_{ma}/dx, (86); dC_{Qi}/dx, (82) and dC_{Hi}/dx, (84). At this stage we are interested only in the first three quantities, which are needed for a balance calculation resulting in θ to trim for each blade.

CALCULATION OF BLADE LOADINGS – FORWARD FLIGHT

AZIMUTH ANGLE ϕ = DEGREES

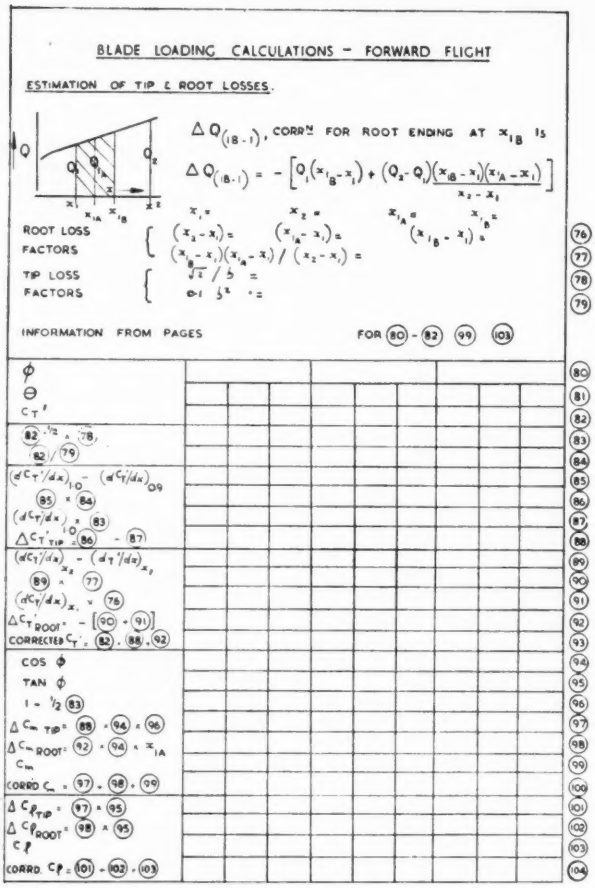
LIFT CURVE SLOPE $(a_o)_t = 2\pi + \epsilon/c\,(4.75 + 0.02\,\epsilon_M)$ =

ϵ/c = ϵ_M = FOR SECTION

DATUM PITCH ANGLE θ_o = RADS.

x	0·2	0·3	0·4	0·5	0·6	0·7	0·8	0·9	1·0	
α_r FROM SHEET A										(43)
$(C_L$ = (39) (43) IF REQD										(44)
dC_T'/dx = (36) (44) /2										(45)
$dC_T'/dx \times \frac{(33)}{C_T'}$										(46)
										(47)
dC_{ℓ_a}/dx = (37) (46)										(48)
C_{ℓ_a}										(49)
										(50)
dC_{m_o}/dx = (38) (46)										(51)
C_{m_a}										(52)
										(53)
DATUM PITCH ANGLE θ_o = RADS										(54)
α_r										(55)
$(C_L$ = (39) (54) IF REQD										(56)
dC_T'/dx = (45) (54) / (43)										(57)
$dC_T'/dx \times \frac{(33)}{C_T'}$										(58)
										(59)
dC_{ℓ_a}/dx = (48) (54) / (43)										(60)
C_{ℓ_a}										(61)
										(62)
dC_m/dx = (51) (54) / (43)										(63)
C_{m_a}										(64)
DATUM PITCH ANGLE θ_o = RADS										(65)
α_r										(66)
$(C_L$ = (39) (65)) IF REQD										(67)
dC_T'/dx = (45) (65) / (43)										(68)
$dC_T'/dx \times \frac{(33)}{C_T'}$										(69)
										(70)
dC_{ℓ_a}/dx = (48) (65) / (43)										(71)
C_{ℓ_a}										(72)
										(73)
dC_m/dx = (51) (65) / (43)										(74)
C_{m_a}										(75)

SHEET B.


BLADE LOADING CALCULATIONS – FORWARD FLIGHT
ESTIMATION OF TIP & ROOT LOSSES.
ROOT LOSS FACTORS
TIP LOSS FACTORS
INFORMATION FROM PAGES
FOR (80) - (82) (99) (103)
SHEET C.

If the area under the curve of dC_T'/dx between 0·9 and 1·0 is assumed to be trapezoidal then

$$\Delta C_{T'\,\mathrm{Tip}} = -\frac{\sqrt{(2C_T')}}{b} \times \left(\frac{dC_T'}{dx}\right)_{B'} \quad . \quad (103)$$

where $(dC_T'/dx)_{B'}$ is mean ordinate at $B' = 1 - \dfrac{\sqrt{(C_T'/2)}}{b}$

Now

$$(dC_T'/dx)_{B'} = (dC_T'/dx)_{1\cdot0} - \frac{\sqrt{(C_T'/2)}}{0\cdot1b}[(dC_T'dx)_{1\cdot0} - (dC'_T/dx)_{0\cdot9}] \quad (104)$$

Hence in (103)

$$\Delta C_{T'\,\mathrm{Tip}} = -(dC_T'/dx)_{1\cdot0}\frac{\sqrt{(2C_T')}}{b} + \frac{C_T'}{0\cdot1b^2}[(dC_T'/dx)_{1\cdot0} - (dC_T'/dx)_{0\cdot9}]. \quad . \quad (105)$$

Because $\Delta C_m = \Delta C_T' x \cos\phi$

then $$\Delta C_{m\mathrm{a\,Tip}} = \Delta C_{T'\,\mathrm{Tip}}\left[1 - \frac{(C_T'/2)^{0\cdot5}}{b}\right]\cos\phi \quad . \quad (106)$$

and similarly

$$\Delta C_{\mathrm{la\,Tip}} = \Delta C_{m\mathrm{a\,Tip}} \tan\phi. \quad . \quad . \quad (106a)$$

Root Losses:

Here the problem has to be treated in a general way to account for the various possible blade-root stations. In general the blade root is at or near $x = 0\cdot25$.

Referring to the sketch on sheet C we solve for a blade-root at $x = x_{1B}$, where $x_{1B} < x = 0\cdot3$, x is generally 0·2, x_2 is generally 0·3.

Let Q be some function such as dC_T'/dx. Then working on the same lines as for tip loss correction:

$$\Delta Q_{1B-1} = -Q_{1A}(x_{1B} - x_1) \quad . \quad . \quad . \quad . \quad (107)$$

and

$$Q_{1A} = Q_1 + (Q_2 - Q_1)\frac{x_{1A} - x_1}{x_2 - x_1}. \quad . \quad . \quad (108)$$

Therefore

$$\Delta Q_{1B-1} = -\left[Q_1(x_{1B} - x_1) + (Q_2 - Q_1)\frac{(x_{1B} - x_1)(x_{1A} - x_1)}{(x_2 - x_1)}\right] \quad . \quad . \quad . \quad (109)$$

Substitution of the appropriate values of (dC_T'/dx) gives

$$\Delta C_{T'\,\mathrm{Root}}.$$

Hence: $$\Delta C_{m\mathrm{a\,Root}} = \Delta C_{T'\,\mathrm{Root}}\, x_{1A}\cos\phi \quad . \quad . \quad (110)$$

and $$\Delta C_{\mathrm{la\,Root}} = \Delta C_{m\mathrm{a\,Root}} \tan\phi. \quad . \quad . \quad (111)$$

The whole of Sheet C is taken up by the solution of the tip and root losses from equations (104)-(111).

The corrected values of C_T', etc., follow from

$$\text{CORRECTED } C_T' = C_T' \text{ line } 82 + \Delta C_T'_{\mathrm{Tip}} + \Delta C_T'_{\mathrm{Root}} \quad . \quad (112)$$

and similar expressions are used for $C_{m\mathrm{a}}$ and C_{la}.

The work that now follows preparatory to Sheet D involves graphical plotting to satisfy the following conditions.

Let $\Sigma C_T'$, $\Sigma C_{m\mathrm{a}}$, ΣC_{la} be the total thrust, pitching moment and rolling moment coefficients derived on Sheet C for three blades at azimuth angles ϕ_n, $\phi_n + 120$, $\phi_n + 240$ degrees, which will be called positions 1, 2 and 3.

If the helicopter is to be in balance then the following must be satisfied.

Vertical Forces:

$$C_T' = \Sigma C_T' = C_T' + C_T'_2 + C_T'_3 = [C_W \cos X + \tfrac{1}{2}C_{\mathrm{DPV}}\lambda_0^2 + \tfrac{1}{2}C_{\mathrm{DPH}}\mu\lambda_0]/\cos\omega. \quad . \quad (29)$$

Pitching Moment:

$$C_m = \Sigma C_{m\mathrm{a}} = \frac{C_W}{R}(x_W \cos\chi - z_W \sin\chi) + 0\cdot5\,(C_{\mathrm{DP}}\mu/R)\,\nu_0^2 z_D + C_{m\mathrm{C.F.}}. \quad . \quad (31a)$$

Rolling Moment:

$$C_l = \Sigma C_{\mathrm{la}} = (C_{\mathrm{TTR}}/R)\,z_t - (C_W/R)(y_W \cos W + z_W \sin W) - 0\cdot5\,(C_{\mathrm{DPV}}/R)\,\nu_0^2 y_D \cos\omega + C_{\mathrm{lC.F.}} \quad . \quad (32a)$$

If $n = 0°$ for example then C_l may be satisfied by considering only θ_2 and θ_3. That is to say it is possible to plot θ_3 in terms of θ_2 to satisfy condition (32*a*).

This then ties θ_2 to θ_3 and the other two conditions

may be solved in terms of (say) θ_2 and θ_1 by reference to this relationship. It is found that there is a unique combination of θ_1, θ_2 and θ_3 which satisfies these conditions provided that the lift-curve slope is assumed constant. This assumption is invalid near the stall but if a slightly lower value of a_0 than the value appropriate to $C_L=0$ is taken, then the only error is a slight pessimism as to the pitch angles required for a given thrust.

Beyond the point at which much of one of a set of blades is stalled it will be found impossible to satisfy all three conditions. Flight may still be possible, however, although considerable vibration would be expected, and the limitation may be that of available cyclic-pitch control.

It is of note that the three conditions (29), (31*a*) and (32*a*) cannot be satisfied by two-bladed rotors, although it is appreciated that this is a common configuration. A full treatment for a two-bladed rotor would involve the inertia and response of the helicopter. This is an argument for three or more blades. In fact the more blades there are, the easier it is to satisfy the conditions.

At this stage it is convenient to plot the required pitch-angles θ so as to obtain their variation around the disc. It will be remembered that in Section 2 when speaking of the dynamics of blade flapping it was mentioned that there was a control advance required to take account of the lag in the system. This advance can be brought into the calculations at this stage if desired, so that the pitch angles θ will be plotted at an angle ϕ in advance of that for which they were calculated.

Advance angles of between 40 and 90 degrees will be encountered depending upon the rigidity of the blades in flapping, blade mass, and so on.

It is found in general that no simple harmonic relationship will closely approximate to the θ variation which may be called $\theta=F_1(\phi)$. A best approximation of this kind, namely

$$\theta=\theta_0+K_1\cos\phi+k_2\sin\phi$$

may be found representing the nearest approach to trim that is possible with a simple swash-plate or spider mechanism. Even the use of second harmonic terms is not enough usually to produce a law sufficiently near $\theta=F_1(\phi)$.

It follows that a device is needed that will monitor the motion put into the blades by the swash-plate or spider mechanism if a minimum of vibration and blade root fatigue is to be attained. Such a device should have an output that is not necessarily harmonic in character and that can be varied at will to suit the various operating tip-speed ratios.

SHEET D

This sheet makes use of the pitch-angle derived above. This angle θ_0 will have to be obtained from the plotted results, including the required control advance. (The term θ_0 used here has already been defined and should not be confused with the θ_0 used in the swash-plate law given in the previous section).

CALCULATION OF BLADE LOADINGS —FORWARD FLIGHT

AZIMUTH ANGLE ϕ = DEGREES

DERIVED PITCH ANGLE Θ_0 = RADS..... FROM SHEET No. (A)

C_{LMAX} AT R.N.=6×10^6 (WINGS 01.01,06) = (B)

KINEMATIC VISCOSITY ALT = FT ΔT = °C = (C)

PROFILE DRAG FACTOR (RA=SD.S. WINGS 02.04.02) =

TRANSITION POINT AT % CHORD MEAN ROUGHNESS HT = FT.

x	0·2	0·3	0·4	0·5	0·6	0·7	0·8	0·9	1·0	
$\Theta_0\cos\delta$ = (A)(18)										(D)
α_T = (34) + (E)										(E)
C_L = (M) (F)										(F)
dC_T'/dx = (33)(36) (G)/2										(G)
C_T'										(H)
$dC_{l\alpha}/dx$ = (37) (H)										(I)
$C_{l\alpha}$										(J)
$dC_{m\alpha}/dx$ = (38) (H)										(K)
$C_{m\alpha}$										(L)
ε + (19)										(M)
$\frac{\omega}{\Omega}$ = (N)/23										(N)
G										(O)
$dC_{Q\alpha}/dx$ = (16)(28)(D)(H)(P)/2										(P)
$C_{Q\alpha}$										(Q)
⊙ GAT x = 0·984										(R)
$dC_{H\alpha}/dx$ = (Q)(18) / (D)(16)										(S)
$C_{H\alpha}$										(T)
										(U)
R.N.=[(7) ΩR(11)] / (B)(15) *										(V)
ΔC_{LMAX} @ R.N.(W) (DA 3176)										(W)
C_{LMAX} @ R.N.(W)										(X)
C_L/C_{LMAX} = (G)/(Y) *										(Y)
$C_{D\delta}-C_{D0}$ (DA 3164)										(Z)
$C_F\times2$ (DO 473 ISS. 2)										(a)
CHORD / ROUGHNESS HEIGHT										(b)
ROUGHNESS FACTOR (WINGS 02.04.08)										(c)
C_{D0} = (b)(d)(C) *										(d)
$C_{D\delta}$ = (e) + (a) +										(e)
dC_{Q0}/dx = [(8)(12)(15)(24)(D)(f)/2										(f)
C_{Q0}										(g)
										(h)
dC_{H0}/dx = (18)(f)/(D)(15)										(i)
C_{H0}										(j)
										(k)
C_Q = (R) + (4)										(l)
										(m)
										(n)

+ IF BLADE SHANK USE EQN(116) : $C_{D\delta}=\cos^3\gamma\,[(t/c)^2+2\,C_F]+0{\cdot}015$

* THESE CALCS BY SLIDE RULE — THESE COLS. ONLY NEED CALCN. FOR C_{D0}

SHEET D.

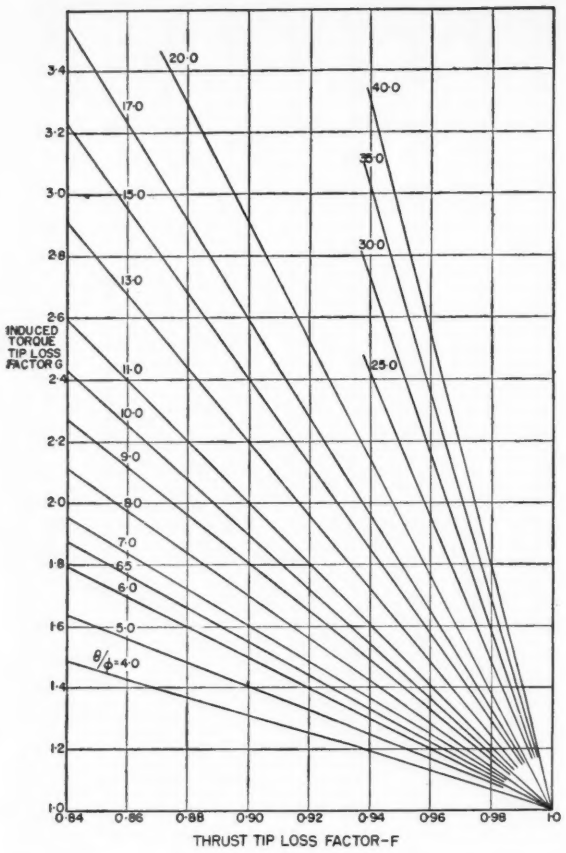
As for previous calculations, C_T', C_l and C_m follow, and the additional parameters are estimated:—

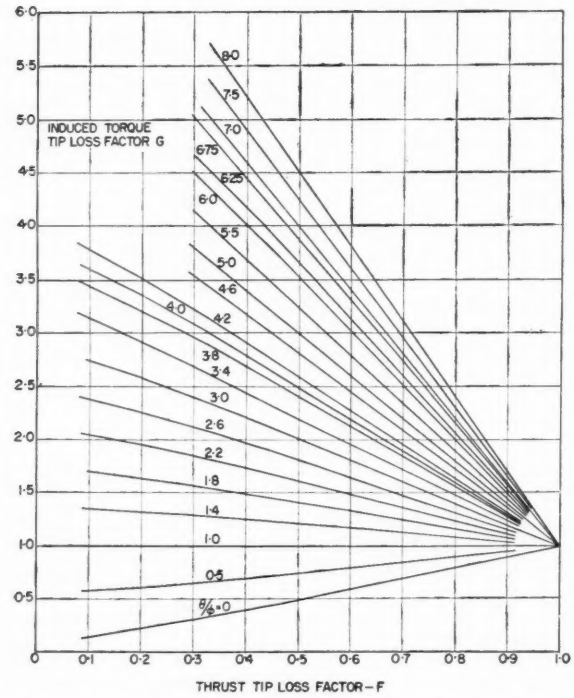
C_{Qi}	from	(82)
C_{Hi}	,,	(84)
C_{Q0}	,,	(83)
C_{H0}	,,	(85)

The tip loss factor G has to be evaluated by the methods of Ref. 15 before C_{Qi} can be determined. This may be done using Figs. 12 and 13 appended to this report.

To obtain C_{Q0} it is necessary to estimate C_{D0}'. The method recommended by Bailey[14] has been found to be most satisfactory and is re-presented (Fig. 14) in a somewhat modified form, suitable only for non-cambered aerofoils.

(*i*) As a first step Reynolds number is evaluated. Now R.N. $=V_e$/Kinematic Viscosity.





FIGURES 12 (*above*) and 13 (*below*). Relation between induced torque and thrust tip loss factors for a three-bladed rotor or airscrew.

$$\frac{\theta}{\phi} = \frac{\text{geometric pitch}}{\text{total induced angle}}$$

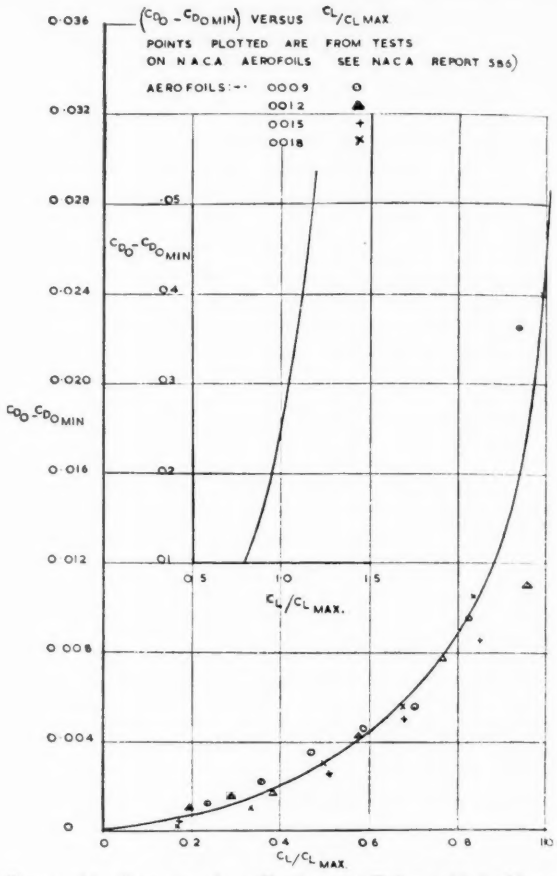


FIGURE 14. Increase of profile drag coefficient with incidence.

Hence in terms of μ'' and taking sweepback into account we derive in line (W):

$$\text{R.N.} = \mu'' \Omega R c / \text{Kin. Visc.} \cos\gamma \quad . \quad (113)$$

(*ii*) $C_{L\,max}$ is read from Data Sheet " Wings," 01.01.06, prepared by the Royal Aeronautical Society or from its derivation[11]. This $C_{L\,max}$ value refers to R.N. $= 6 \times 10^6$ and requires correcting to actual R.N. This may be done by " Wings 01.01.06 " or by Fig. 15 which extends the range of this information based on tests on N.A.C.A. 4-digit series aerofoils[12]. This leads to $C_{L\,max}$ at R.N. = (W).

The value of $C_{L\,max}$ so derived is independent of θ_0, and therefore should have been used to check on the values of C_i worked out earlier on Sheet B.

(*iii*) The ratio $C_L/C_{L\,max}$ is used to derive the increase in sectional (*i.e.* 2-dimensional) profile drag due to lift. In Ref. 3 the expression for C_{D_0}' is given as $C_{D_0}' = C_{D_0} + \delta_1 z_r + \delta_2 z_r^2$, but is thought not to cater adequately for Reynolds number effects. The method used by Bailey[14] is preferred and Fig. 14 gives $C_{D_0}' - C_{D_0}$ versus $C_L/C_{L\,max}$ based on the Wind Tunnel tests of Ref. 12. The inset extension diagram takes this function up to

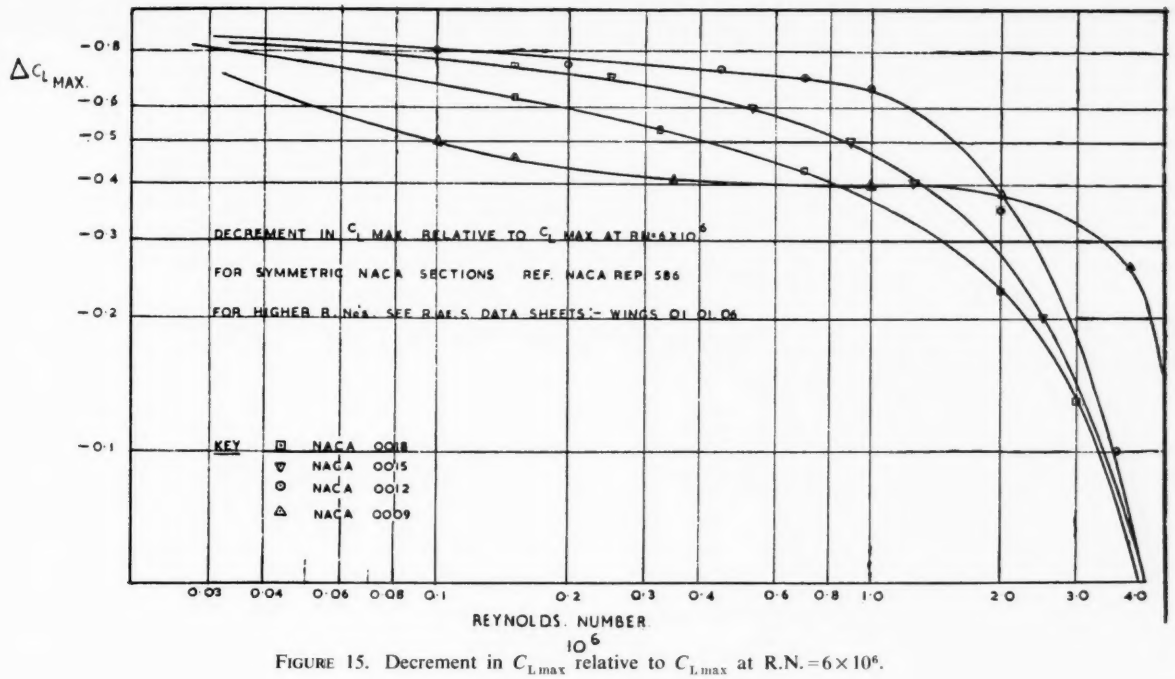


FIGURE 15. Decrement in $C_{L\,max}$ relative to $C_{L\,max}$ at R.N. $=6\times10^6$.

$C_L/C_{L\,max}=1{\cdot}2$, based on an extrapolation giving the drag increment approximately proportional to the cube root of the C_L ratio.

(*iv*) Various methods may be used to estimate C_F. Fig. 16 is based upon Falkner's law for turbulent flow for R.N. $<5\times10^6$, and agrees well with Royal Aeronautical Society Data Sheet "Wings 02.04.01."

The variation of C_F with R.N. and the effects of thickness/chord ratio and roughness on C_{D_0} mean that no unique value can be assumed as is done in Ref. 3 where $C_{D_0}=0{\cdot}0087$ is recommended.

The t/c ratio effect is given in Royal Aeronautical Society Data Sheet "Wings 02.04.02," *see* line (C), or the expression in Ref. 13 may be used:

$$C_{D_0}/(2C_F)=1+2t/c+60\,(t/c)^4. \quad . \quad (114)$$

Lines (c) and (d) give the two factors on C_F and are used to obtain C_{D_0} in line (e). Hence we obtain C_{D_0}' in line (f).

Where there is a blade shank it is necessary to estimate the drag of an inclined cylinder. The drag of such a cylinder is given by:

$$C_{D_0}'\simeq\cos^3\gamma\,[(t/c)^2+2C_F]+0{\cdot}015 \quad . \quad (115)$$

this expression being taken from Hoerner[13].

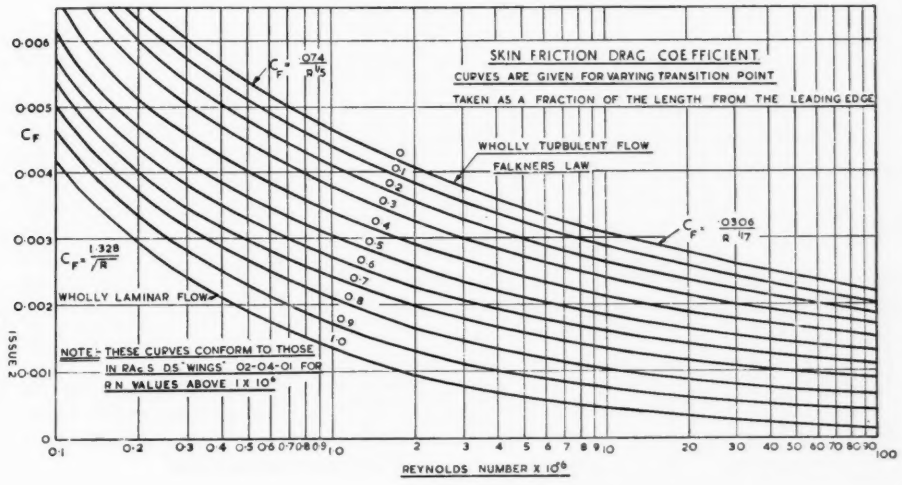


FIGURE 16.
Skin friction drag coefficient.

BLADE LOADING CALCULATIONS — FORWARD FLIGHT
COEFFICIENTS CORRECTED FOR TIP & ROOT LOSSES

INFORMATIONS FROM SHEET D, PAGES . FOR ϕ = ETC.
& FROM FOR ϕ = ETC.

x_B = x_{1A} = x_1 =

ϕ							105
θ FROM (A)							106
C_T' FROM (I)							107
C_e FROM (L)							108
C_{m_0} FROM (O)							109
C_{Q_0} FROM (R)							110
C_{H_0} FROM (U)							111
$(dC_T'/dx)_{1\cdot0} - (dC_T'/dx)_{0\cdot9}$							112
(84)* × (112)							113
$(dC_T'/dx)_{1\cdot0}$ × (83)*							114
$\Delta C_{T'TIP}$ = (113) − (114)							115
$(dC_{T'}/dx)_{x_{1B}} - (dC_{T'}/dx)_{x_1}$							116
(77) × (116)							117
$(dC_{T'}/dx)_{x_1}$ × (76)							118
$\Delta C_{T'ROOT}$ = [(117) × (118)]							119
CORR^{n} $C_{T'}$ = (107) + (115) + (119)							120
COS ϕ							121
TAN ϕ							122
$1-\frac{1}{2}$ (83)*							123
$\Delta C_{m\ TIP}$ = (115) × (121) × (123)							124
$\Delta C_{m\ ROOT}$ = (119) × (121) × x_{1B}							125
CORR^{n} C_m = (109) + (124) + (125)							126
$\Delta C_{e\ TIP}$ = (124) × (122)							127
$\Delta C_{e\ ROOT}$ = (125) × (122)							128
CORR^{n} C_e = (108) + (127) + (128)							129
$(dC_Q/dx)_{1\cdot0} - (dC_{Q_0}/dx)_{0\cdot9}$							130
(84)* × (130)							131
$(dC_{Q_0}/dx)_{1\cdot0}$ × (83)*							132
$\Delta C_{Q_0 TIP}$ = (131) − (132)							133
$(dC_Q/dx)_{x_1} - (dC_{Q_0}/dx)_{x_1}$							134
(77) × (134)							135
$(dC_{Q_0}/dx)_{x_1}$ × (76)							136
$\Delta C_{Q_0 ROOT}$ = − [(135) + (136)]							137
CORR^{n} C_{Q_0} = (110) + (133) + (137)							138
$1-\frac{1}{2}$ × (83)*							139
γ_1* APPROPRIATE TO (139)							140
COS γ_1							141
SIN $(\gamma_1+\phi)$							142
$\Delta C_{H_0 TIP}$ = (133) × (142) / (139) × (141)							143

84* NOTE. ITEMS THUS* ARE DERIVED FROM INTERPOLATION ON APPROPRIATE SHEETS.

SHEET E.

SHEET F.

BLADE LOADING CALCULATIONS — FORWARD FLIGHT
COEFFICIENTS CORRECTED FOR TIP & ROOT LOSSES
WORK CONTINUED FROM SHEET E:
TORQUE FACTOR, ν ETC. FROM WORK REF.

FOR C_{Q_0} & C_{H_0} ROOT LOSSES ONLY. TAKE x_1' = x_1' = x_{1_A}' = x_{1_B}' = 0
$(x_2'-x_1')$ = $(x_{1_A}'-x_1')$ = $(x_{1_b}-x_1')$ = (1)
$(x_{1_B}'-x_1')(x_{1_A}'-x_1')/(x_2'-x_1')$ = (152)

ϕ							
γ_2 APP. TO x_{1A}							144
COS γ_2							145
SIN $(\gamma_2+\phi)$							146
$\Delta C_{H_0 ROOT}$ = (137) × (146) / x_{1_A} × (145)							147
CORR^{n} C_{H_0} = (111) + (143) + (147)							148
C_{Q_0} FROM (N)							149
C_{H_0} FROM (N)							150
$(dC_{Q_0}/dx)_{x_2'} - (dC_{Q_0}/dx)_{x_1'}$							153
(152) (153)							154
$(dC_{Q_0}/dx)_{x_1'}$ × (151)							155
ΔC_{Q_0} = − [(154) + (155)]							156
CORR^{n} C_{Q_0} = (149) + (156)							157
γ_2' APP. TO x_{1_A}'							158
COS γ_2'							159
SIN $(\gamma_2'+\phi)$							160
ΔC_{H_0} = (56) × (160) / x_{1_A}' × (159)							161
CORR^{n} C_{H_0} = (150) + (161)							162
TOTAL C_H = (148) + (162)							163
ΣC_H							164
TOTAL C_Q = (138) + (157)							165
ΣC_Q							166
TORQUE $\Phi = (\Sigma C_Q)\pi\eta^3\rho(\Omega R)^2$			–		–		167
NETT B.H.P. REQ^{D} = ΩQ/550			–		–		168

3.6. SHEET E

This is identical in most respects to the method of Sheet C, being a correction sheet for tip and root losses to the coefficients affected directly by lift. As in Sheet C, if tip losses have been estimated by the method of Ref. 15 then the following lines are omitted:

(112)-(115), (124), (127), (130)-(133).

The sheet is laid out so as to collect information for two sets of azimuth positions for a three-bladed rotor.

3.7. SHEET F

The profile torque is not assumed to suffer from tip effects, and the full incidence contribution to C_{D_0} is retained.

Now

$$C_T \propto (\theta' - \phi') \longrightarrow 0 \text{ at } x = 1{\cdot}0$$

$$C_{Q_1} \propto (\theta' - \phi')\,\phi \longrightarrow 0 \text{ at } x = 1{\cdot}0$$

because $\phi' \longrightarrow \theta'$ at $x = 1{\cdot}0$.

It follows that C_{Q_1} approaches the limit in a manner dissimilar to C_T, and that the equivalent limit of integration is greater than B.

The root effect present on this sheet attempts to approximate to the shank drag. Although the torque contribution of this part is small there is a larger effect on the H force. For this purpose therefore the position of x_{1B} (*see* Sheet C) is made zero. (Note the symbols x_{1B}, etc., are given a superscript to denote application to C_{Q_0} and C_{H_0} only).

The final lines of this calculation check back upon the B.H.P. required at the rotor. Agreement will only be obtained with the value of power available inserted into the performance equation (15) in Section 1 if the third term in (15) is correct both in δ and the coefficient of ν_0^2, and also in the approximation to C_H in (9) which is used in (21). This approximation to C_H can also be checked.

Work carried out to date shows that the value of δ is approximately correct if calculated as detailed in Section 1. The approximation to C_H appears also to be not unreasonable, a fact which is supported by analysis of the wind tunnel tests on a 12 ft. diameter rotor described in Ref. 16. The coefficient of ν_0^2, usually given as 4·5 was found to be low and a value of 6·0 fitted the two flight conditions investigated.

It was found that there is a significant variation in torque with time, *i.e.* during one revolution of the rotor, on the three-bladed rotor system that has been investigated. It would appear to be worthwhile increasing the number of blades to a minimum of four if this fluctuation (of the order of 70 per cent.) is thought to be undesirable.

CALCULATION OF BLADE LOADINGS IN FORWARD FLIGHT.
EFFECT OF GUSTS.

A.U.W. LB. Ω_R = FT/SEC. (N)
ENGINE CONDITION :— R.P.M. BOOST
AZIMUTH ANGLE. = DEGS. TRIMMED BLADE PITCH = RADS

α_R FROM SHEET D	(F)									
GUST VELOCITY =	F.P.S. = ω_1									
$\Delta\alpha_\gamma = \tan^{-1}\omega_1/\Omega_R = \omega_1/$ (11)(A)	(B)									
α_{TPn} = (F) + (B)	(T)									
C_L = (E) (39)										
$dC_T'/d\chi$ = (H)(L)/(F)	(V)									
$C_T'n$										
$n = C_T'n / C_T'$										
$dC_L/d\chi$ = (V) (37)										
C_L										
$dC_m/d\chi$ = (V) (38)										
C_m										
GUST VELOCITY =	F.P.S. = ω_2									
$\Delta\alpha_\gamma = \tan^{-1}$ (B)(ω_2)/(ω_1)	(W)									
α_{TPn} = (F) + (W)	(Y)									
C_L = (Y) (39)										
$dC_T'/d\chi$ = (V)(Y)/(L)	(Z)									
$C_T'n$										
$n = C_T'n / C_T'$										
$dC_L/d\chi$ = (Z) (37)										
C_L										
$dC_m/d\chi$ = (Z) (38)										
C_m										
GUST VELOCITY =	F.P.S. = ω_3									
$\Delta\alpha_\gamma = \tan^{-1}$ (B)(ω_3)/(ω_1)	(P)									
α_{TPn} = (F) + (P)	(E)									
C_L = (E) (39)										
$dC_T'/d\chi$ = (V)(E)/(L)	(π)									
$C_T'n$										
$n = C_T'n / C_T'$										
$dC_L/d\chi$ = (π) (37)										
C_L										
$dC_m/d\chi$ = (π) (38)										
C_m										
$x = r/R$		0·2	0·3	0·4	0·5	0·6	0·7	0·8	0·9	1·0

SHEET G.

3.8. SHEET G

This sheet is used to obtain the loadings due to gusts. It can also be used to obtain the rolling and pitching moments and average acceleration on the aircraft.

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REFERENCES

1. SQUIRE, H. B. The Flight of a Helicopter. R. & M. 1730.
2. GLAUERT, H. A General Theory of the Autogiro. R. & M. 1111.
3. GESSOW, A. and MYERS, G. C. *Aerodynamics of the Helicopter.* Macmillan.
4. WALD, Q. (1943). Rapid Estimation of Helicopter Performance. *Journal of the Aeronautical Sciences.* April 1943.
5. O'HARA, F. A Method of Performance Reduction for Helicopters. A.F.E.E. Res. 26.
6. KEITH-LUCAS, D. (1952). The Shape of Wings to Come. British Association Meeting, Belfast, September 1952.
7. MEYER DREES, J. A Theory of Airflow through Rotors and its Application to some Helicopter Problems. *Journal of Helicopter Association of Great Britain,* Vol. 3, No. 2.
8. COLLINGBOURNE, J. R. The Estimation of Lift-Curve Slope at Subsonic Mach Numbers. Tech. Note Aero. 2145.
9. WEBER, J. and BREBNER, G. G. A Simple Estimate of the Profile Drag of Swept Wings. Tech. Note Aero. 2168.
10. LOCK, C. N. H. and YEATMAN, D. Tables for Use in an Improved Method of Airscrew Strip Theory Calculations. R. & M. 1674.
11. MULTHOPP, H. On the Maximum Lift Coefficient of Aerofoil Sections. A.R.C. 12, 115.
12. JACOBS, E. N. and SHERMAN, A. Aerfoil Section Characteristics as Affected by Variations of Reynolds Number. N.A.C.A. Report 586.
13. HOERNER, S. F. Aerodynamic Drag. Otterbein Press.
14. A Simplified Theoretical Method of Determining the Characteristics of a Lifting Motor in Forward Flight. N.A.C.A. Report 716.
15. MARSHALL, J. Investigation into the Effects of Tip-losses on the Thrust and Torque Loadings on a Helicopter Rotor in Forward Flight. S. B. & H. Ltd. Aerodynamics Report No. 69.
16. SQUIRE, H. B., FAIL, R. A. and EYRE, R. C. W. Wind-tunnel Tests on a 12 ft. Diameter Helicopter Rotor. Report Aero. 2324.

APPENDIX I

Expression for C_T'

Equation (6) may be rewritten:—

$$\cos\chi = \left[1 - \frac{0{\cdot}25\mu^2}{C_W{}^2}(\sigma\delta + C_{DPH}\mu)^2\right]^{0{\cdot}5}$$

$$\simeq \left[1 - \frac{0{\cdot}125\mu^2}{C_W{}^2}(\sigma\delta + C_{DPH}\mu)^2\right]. \qquad (6a)$$

And (7) gives)

$$C_T' = C_W \cos\chi + 0{\cdot}5C_{DPV}\lambda_0{}^2. \quad . \quad . \quad (7a)$$

Therefore

$$C_T' = C_W - \frac{0{\cdot}125\mu^2}{C_W}(\sigma\delta + C_{DPH}\mu)^2 + 0{\cdot}5C_{DPV}\lambda_0{}^2$$

$$= C_W - \frac{0{\cdot}125\,(v_0{}^2 - \lambda_0{}^2)}{C_W}[(\sigma\delta)^2 + 2C_{DPH}\mu\sigma\delta + C_{DPH}{}^2\mu^2] + $$
$$+ 0{\cdot}5C_{DPV}\lambda_0{}^2. \quad . \quad . \quad (a)$$

Now $\sigma\delta$ is small compared with $C_{DPH}\mu$. Hence (*a*) may be approximated by

$$C_T' \simeq C_W - (0{\cdot}125C_{DPH}{}^2v_0{}^4/C_W) + (0{\cdot}25C_{DPH}{}^2v_0{}^2\lambda_0{}^2/C_W) + $$
$$+ 0{\cdot}5C_{DPV}\lambda_0{}^2 - (0{\cdot}125C_{DPH}{}^2\lambda_0{}^4/C_W) \quad . \quad (b)$$

and this may be further simplified to

$$C_T' \simeq C_W + 0{\cdot}5C_{DPV}\lambda_0{}^2 \quad . \quad . \quad . \quad (c)$$

i.e. cos χ may be replaced by 1·0.

Therefore

$$C_T'{}^2 \simeq C_W{}^2 + C_{DPV}\lambda_0{}^2.$$

Expression for $\overline{\Delta}$

The Momentum Theory gives values for $\overline{\Delta}$ (*see* equation (11)) which are valid for v_0 not near zero. Empirical evaluation of $\overline{\Delta}$ for hovering flight gives a factor = 1·2 (*see example* Refs. 3 and 5). The following modification to the Momentum Theory expression is suggested:—

$$\overline{\Delta} = (1{\cdot}2 - v_0{}^2)\,C_T'/2v_0 \quad . \quad . \quad . \quad (d)$$

APPENDIX II

TABLE I

FROM EQUATION (90) $\mu'/\mu = \cos(i+\beta\cos\phi)/\cos i$, $\beta = 0{\cdot}1$.

ϕ	i	0	0·05	0·10	0·15	0·20
0°	360°	0·995,00	0·990,01	0·985,59	0·979,92	0·974,77
30°	330°	0·996,249	0·991,88	0·987,57	0·983,47	0·978,72
60°	300°	0·998,750	0·996,24	0·993,73	0·991,81	0·998,82
90°	270°	1·000,000	1·000,00	1·000,00	1·000,00	1·000,00
120°	240°	0·998,750	1·001,25	1·003,76	1·006,32	1·008,88
150°	210°	0·996,249	1·000,69	1·004,93	1·009,31	1·013,77
180°		0·995,00	1·000,00	1·005,02	1·010,09	1·015,24

TABLE II

FROM EQUATION (91) $K'/\mu = \sin(i+\beta\cos\phi)/\cos i$.

ϕ	i	0	0·05	0·10	0·15	0·20
0°	360°	0·100,34	0·149,63	0·197,68	0·250,21	0·301,61
30°	330°	0·086,55	0·136,35	0·186,36	0·237,06	0·288,47
60°	300°	0·049,98	0·100,47	0·150,19	0·198,92	0·252,43
90°	270°	0	0·050,04	0·100,84	0·151,14	0·200,69
120°	240°	−0·049,98	0	0·050,23	0·101,48	0·152,48
150°	210°	−0·086,55	−0·034,04	0·013,57	0·064,17	0·115,56
180°		−0·100,34	−0·050,004	0	0·050,55	0·102,38

TABLE III

FROM EQUATION (92) $\mu''/\mu' = [(x/\mu')^2 + 1 + (2x/\mu')\sin\phi]^{0{\cdot}5}$.

ϕ \ x/μ'	0·5	1·0	2·0	4·0	8·0	18
0°	1·118,07	1·414,22	2·236,07	4·123,25	8·062,26	16·031,2
30°	1·322,85	1·732,05	2·645,75	4·582,60	8·544,00	
60°	1·454,65	1·931,85	2·909,32	4·891,16	8·880,12	16·873,4
90°	1·500,00	2·000,00	3·000,00	5·000,00	9·000,00	17·000,0
120°	1·454,65	1·931,85	2·909,32	4·891,65	8·880,12	16·873,4
150°	1·322,55	1·732,05	2·645,75	4·582,60	8·544,00	
180°	1·118,07	1·414,22	2·236,07	4·123,25	8·062,26	16·031,2
210°	0·866,03	1·000,00	1·732,05	3·605,60	7·549,80	15·524,5
240°	0·619,65	0·517,63	1·293,07	3·173,60	7·289,96	15·142,2
270°	0·500,00	0	1·000,00	3·000,00	7·000,00	15·000,0
300°	0·619,65	0·517,63	1·293,07	3·173,60	7·289,96	15·142,2
330°	0·866,03	1·000,00	1·732,05	3·605,60	7·549,80	15·524,5

TABLE IV

FROM EQUATION (97) $\gamma = \sin^{-1}[\mu'\cos\phi/\mu'']$.

ϕ	μ'/μ''	1·0	0·75	0·5	0·25	0·125
0°	360°	90	48·6	30	14·48	7·32
30°	330°	60	40·5	25·56	12·50	6·22
60°	300°	30	22·0	14·48	7·32	3·58
90°	270°	0	0	0	0	0
120°	240°	−30	−22·0	−14·48	− 7·32	−3·58
150°	210°	−60	−40·5	−25·56	−12·50	−6·22
180°		−90	−48·6	−30·00	−14·48	−7·32

APPENDIX III

Blade Tip-losses in Forward Flight

The three-dimensional effect that is termed "tip-loss" operates on the local induced velocity and hence on local induced angle ϕ'. The value of ϕ has to be increased in such a way that as $r \longrightarrow R$, $\phi' \longrightarrow \theta'$, $T' \longrightarrow 0$ (*i.e.* $L \longrightarrow 0$).

Now $Q_i \propto (\theta - \phi')\phi'$ so that although $Q_i \longrightarrow 0$ as $r \longrightarrow R$, the increasing values of ϕ' ensure that the limit will be approached in a manner radically different to that for T.

It is shown in Ref. 15 that the effects of these changes on T and Q_i are such as to increase Q_i near the tip for large values of θ', but to reduce T for all values of θ'.

Let F = correction factor on thrust . . (1)

then $F = (\theta' - \phi_c')/(\theta' - \phi')$ . . . . (1)

where suffix c indicates three-dimensional value.

If G' = correction factor to induced torque

$$G = (\theta' - \phi_c')\,\phi_c'/(\theta' - \phi')\,\phi' \quad . \quad . \quad . \quad (2)$$

Hence substituting from (1) in (2)

$$G' = \frac{F}{\phi}[\theta - F(\theta - \phi)]$$

or $$G' = \frac{F\theta}{\phi}(1 - F) + F^2. \quad . \quad . \quad . \quad . \quad (3)$$

If we write $G' = G \times F$ . . . (4)

then $G = (\theta/\phi)(1 - F) + F$ . . (5)

or if we write θ and ϕ in terms of the parameters used in Section 2 of this report:

$$G = (1 - F)\,[(\theta_0 + \Delta\theta)\cos\gamma/\{(\Delta + K')/\mu''\}] + F. \quad (6)$$

These tip-loss factors have been calculated in Ref. 15 and Fig. 10 and Figs. 12 and 13 from this reference are appended to this report.

TECHNICAL NOTES

Contributions to this Section of the JOURNAL *will be eligible for Journal Premium Awards and will normally be published within two months of being received.*

A Note on the China-Film Technique for Boundary Layer Indication

by

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THE "CHINA-FILM" technique was developed by Murphy and Phinney[1] from the "China Clay" technique for indicating transition originated by Richards and Burstall[2]. It consists in applying a suspension of china clay in a volatile liquid to a surface subsequently subjected to air flow. Not only is transition indicated, but when all the liquid has evaporated, the china clay remains deposited on the surface in patterns which may be interpreted to yield information on the direction of air flow near the surface and the incidence of shocks thereon.

Experiments were carried out, at supersonic and subsonic air speeds, to determine the process by which the patterns form and to establish the aerodynamic meaning of the "flow lines" of the pattern. The influence of the viscosity of the carrier liquid was also studied. Indications of flow direction were found to be unreliable near a trailing edge, on a vertical surface at low speeds and in regions where the surface shear stress is only of the same order of magnitude as the product of pressure gradient and film thickness.

Except where otherwise stated, the indicator solution used was composed of 1 gm. of china clay in 2 cc. of methyl salicylate.

BOUNDARY LAYER FLOW LINES

It is known[1] that when the liquid has evaporated, streaks of china clay remain forming a pattern of "flow lines" on the surface.

In the present experiments, the formation of these streaks was observed to be connected with the presence of a thin sub-layer of sludge of a thick, muddy consistency, composed of china clay in high concentration in the carrier liquid near the surface. Wavelets with crest transverse to the direction of air flow were present at the liquid-air interface, and when the film of indicator liquid covering the sludge became very thin, embankments of sludge began to build up at the downstream faces of these wavelets. The embankments were limited in extent and the liquid flowed round their ends to form fine filaments which forced a slightly meandering passage through the sludge, but which maintained a mean direction dictated by the action of the air flow. Continual coalescence and bifurcation of the filaments was observed, but the former process greatly outweighed the latter, so that the filaments increased in width and, being then apparently less subject to deflection by the sludge, meandered less from their mean direction. The sludge was removed from the filament paths, formed into embankments along the edges of the filaments and, after completion of the evaporation, appeared as streaks of china clay.

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The china clay pattern may be enhanced by increasing the viscosity of the carrier liquid. Not only are the streaks intensified, due to the greater quantity of indicator retained on the surface, but also the tendency to meander of the fine filaments is increased, so that larger filaments form more rapidly and the prominent streaks of the pattern begin to occur farther upstream.

The interpretation of the pattern is assisted by the following consideration based on the boundary layer approximation. When no liquid is present the direction of the surface shear stress, τ_F, is the same as that of $\partial \mathbf{v}/\partial y$, where $\mathbf{v}$ denotes the velocity vector and y denotes distance normal to the surface, and hence is the same as the limiting direction of the velocity at the surface.

When a thin film of liquid is present, the liquid velocities are small and hence the equations of motion for the liquid reduce approximately to

$$\text{grad } p = \partial \tau / \partial y, \quad . \quad . \quad . \quad (1)$$

$$\partial p / \partial y = 0, \quad . \quad . \quad . \quad (2)$$

where p and τ denote respectively the pressure and the shear stress. Now, $\tau = \mu \partial \mathbf{u}/\partial y$, where μ and $\mathbf{u}$ denote respectively the viscosity and the velocity of the liquid, so by (1) and (2)

$$\mathbf{u} = [\tau_i - (y_i - y/2) \text{ grad } p]\, y/\mu,$$

where the subscript i denotes values at the liquid-air interface. Thus if

$$| \tau_i / (y_i \text{ grad } p) | >> 1 \quad . \quad . \quad (3)$$

then

$$\mathbf{u} = (y/\mu)\, \tau_i; \quad . \quad . \quad . \quad (4)$$

that is, the liquid filaments and china clay streaks follow the direction of the air shear at the liquid-air interface.

Since the liquid film is very thin when the indication starts to form, it appears plausible to assume that the air boundary layer is affected to a negligible degree by its presence and

$$\tau_F \simeq \tau_i. \quad . \quad . \quad . \quad . \quad (5)$$

It then follows that, subject to (3), the direction indicated by the china clay streaks is identical with the direction of surface shear and the limiting direction of the velocity when the indicator is absent.

This result was confirmed experimentally by comparing the direction indicated by the streaks with that indicated by a tuft at the tip of a fine probe lowered into a boundary layer obtained at an air speed of 100 ft./sec. and Reynolds number of 4×10^5. The direction of the streaks (obtained using water as carrier fluid) was almost at a right angle to the direction of pressure gradient and at an angle of approximately 45° to the mainstream velocity direction.

FIGURE 1. China-film indication.

The deviation of the tuft from the mainstream direction increased during approach to the surface and, when the probe tip touched the surface, the direction of the tuft was identical with that of the streaks.

By (5), (3) may be written

$$| c_f/(y_i \text{ grad } c_p) | >> 1$$

where c_f and c_p denote respectively the coefficients of skin friction and pressure. This condition is always satisfied when the liquid layer is very thin and the pressure gradient is not large compared with the local skin friction. But, near separation or shock boundary layer interaction, these requirements may not be met, and in such circumstances the significance of the streak direction is more difficult to determine.

It should also be noted that the tendency of the liquid to accumulate at a trailing edge often distorts or obliterates the china clay pattern there, and it has been found that at low air speeds the indication on vertical surfaces may be affected markedly by the influence of gravity.

Figure 1 is a photograph of an indication obtained by the method in a small, intermittent, supersonic tunnel; the true width of the surface was $\frac{1}{2}$ in. The large divergence of the china clay streaks at the downstream end is probably due to separation caused by an air leak along the side wall at AB. The discontinuity in the china clay deposit at X denotes the trace, on the floor of the tunnel, of a shock arising from the swept leading edge of a model spanning the working section parallel to the floor.

TRANSITION

Considerable study has already been devoted to this aspect[1,2]. It is worth noting, however, that the evaporating time (before indication) is influenced by the viscosity of the carrier liquid. If the viscosity is high, much of the indicator will be retained on the surface in spite of the shear action of the air flow. Not only does the very quantity of liquid then present increase the evaporating time, but the latent heat used in evaporating the liquid nearest the liquid-air interface is extracted from the remaining liquid, so that, unless heat is supplied from the surface, the consequent cooling extends the evaporating time even further.

ACKNOWLEDGMENTS

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REFERENCES

1. MURPHY, J. S. and PHINNEY, R. E. (1951). Visualization of Boundary Layer Flow. Readers' Forum. *Journal of the Aeronautical Sciences*, Vol. 18, p. 771. 1951.
2. RICHARDS, E. J. and BURSTALL, F. H. (1945). The "China Clay" Method of Indicating Transition. R. & M., Aero. Res.Coun. No. 2126, 1945.
3. STALKER, R. J. A Study of the China-Film Technique for Flow Indication. Aeronautical Research Laboratories, Melbourne. Report A.96.

Further Note on the Treatment of Structural Discontinuities

by

W. J. GOODEY

A PROCEDURE was discussed in Ref. 1 for dealing with structural discontinuities by making the strain energy of the structure a minimum, subject to the conditions that the members to be cut out are entirely free from load, and the proof was based on the well-known method of Lagrangian Multipliers. Here is another proof. As in Ref. 1,

$$dU = 2d\psi'(A\psi + BP) = 0. \quad . \quad . \quad (1)$$

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Suppose there are N redundancies in the un-cut structure, this number being reduced to $N - n$ by imposing n conditions

$$F\psi + GP = 0. \quad . \quad . \quad . \quad . \quad (2)$$

The left hand side of (2) simply contains the expressions for the loads in the members to be cut out. Thus F is, in general, a rectangular matrix with n rows and N columns, though in practice most of the columns will contain only zeros. An important point to note here, which was not

mentioned in Ref. 1 probably because it was taken for granted, is that the rows of F must be linearly independent, since otherwise the matrix $FA^{-1}F'$ is singular. Consider, for example, a cut-out involving the removal of two adjacent skin panels and the portion of stringer in between. The removal of the stringer gives two linearly independent conditions (load = 0 at each end). The removal of the skin panels gives two more conditions (shear loading = 0 in each) which are not, however, linearly independent of the other two, since the stringer cut-out has the effect of equalising these shear loadings. Hence there are only *three* linearly independent conditions imposed by the cut-out. When the cut-out is extensive this point may require some thought.

In view of the above, it is possible to construct an $N \times N$ non-singular matrix $\begin{bmatrix} F \\ R \end{bmatrix}$ by the addition to F of a further $N-n$ linearly independent rows, denoted by R.

Now let $$R\psi = \phi. \quad . \quad . \quad . \quad . \quad . \quad (3)$$

Then from (2) and (3)

$$\psi = \begin{bmatrix} F \\ R \end{bmatrix}^{-1} \begin{bmatrix} -GP \\ \phi \end{bmatrix} = -\alpha GP + \beta\phi, \text{ say,}$$

where $[\alpha, \beta]$ is the reciprocal of $\begin{bmatrix} F \\ R \end{bmatrix}$, α being the first n columns. . . . . (4)

The $N-n$ quantities in the column ϕ may be taken as the unknowns for the modified structure. Then, from (1) and (4), $dU = 2d\phi'\beta'(A\psi + BP)$, and since the unknowns are linearly independent the condition for minimum strain energy is

$$\beta'(A\psi + BP) = 0. \quad . \quad . \quad . \quad . \quad (5)$$

Equations (2) and (5) could be solved for ψ, but this involves the preliminary determination of a matrix β, which in turn depends on the choice of R. These steps may be obviated by pre-multiplication of equation (5) by R', and using the relation $F'\alpha' + R'\beta' = N \times N$ unit matrix, whence

$$A\psi + BP = F'\alpha'(A\psi + BP),$$

or $$A\psi + BP + F'\lambda = 0 \quad . \quad . \quad . \quad (6)$$

where $$\lambda = -\alpha'(A\psi + BP)$$

Combining (2) and (6)

$$\begin{bmatrix} A, & F' \\ F, & 0 \end{bmatrix} \begin{bmatrix} \psi \\ \lambda \end{bmatrix} + \begin{bmatrix} BP \\ GP \end{bmatrix} = 0 \quad . \quad . \quad (7)$$

This is the same as equation (8) of Ref. 1.

REFERENCE

1. GOODEY, W. J. (1955). Notes on a General Method for the Treatment of Structural Discontinuities. *Journal of the Royal Aeronautical Society*, October 1955.

The Estimation of the Lift due to Full and Part Span Controls

by

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THIS NOTE DESCRIBES an investigation of experimental and theoretical data on the lift increment due to the deflection of full and part span controls. For full span controls, existing Data Sheets in the Royal Aeronautical Society Aerodynamics series are found to be inadequate for some of the plan forms currently in use, but the investigation shows that with one additional set of curves the sheets may be applied to almost any plan form. For part span conversion factors, a revision also is proposed.

NOTATION

A	wing aspect ratio
$\Lambda_{\frac{1}{4}}$	sweepback of wing quarter-chord line
λ	wing taper ratio
b	wing span
b_f	span of inboard part span control
$\eta = b_f/b$	
c	local wing chord
c_f	local control chord, aft of hinge line
a_1	slope of lift coefficient curve with incidence
a_2	slope of lift coefficient curve with control deflection
ΔC_L	lift coefficient increment due to control deflection
M	Mach number

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Suffices 0 and 6 are used to refer to two-dimensional conditions and to rectangular wings of aspect ratio 6, respectively.

INTRODUCTION

In the Royal Aeronautical Society's Data Sheets in the Aerodynamics series[1], there are a number of sheets dealing with the lift increment due to the deflection of full or part span trailing edge controls. The range of design of control to which the sheets are applicable is naturally limited by the amount of suitable data available for each type, and for this reason the sheets deal only with plain controls, split flaps, and certain types of slotted and double flaps.

These Data Sheets were produced several years ago when there was very little experimental data on wings of low aspect ratio or with appreciable amounts of sweep. Interest at this time was still mainly focused on the characteristics of unswept wings of what would now be considered moderate or high aspect ratio, and accordingly the sheets either referred to rectangular wings of aspect ratio 6, or covered two-dimensional conditions. The effects of compressibility were not considered. Conversion of lift increments determined from these sheets to aspect ratios either larger than, or about, 6 could be made quite accurately by using as a conversion factor the ratio $a_1/(a_1)_6$ or $a_1/(a_1)_0$ as appropriate.

For wings of low aspect ratio or with large amounts of sweep, the use of such conversion factors is clearly open

to doubt and comparison with experiment has shown that for wings of this type the sheets do not give entirely reliable results. Thus if the Data Sheets are to be more generally useful the data must be in some way extended to cover the wide range of wing plan forms currently in use.

Theoretical methods which are applicable to swept wings and wings of low aspect ratio are now available for the determination of control lift effectiveness. Among these can be mentioned the Multhopp lifting surface theory[2] and a simplified Weissinger lifting surface theory as applied by DeYoung[3]. The former of these is rather too long for routine calculations, and for this reason the number of solutions available is strictly limited. The latter method leads to rather simpler computation, the results of which DeYoung presents in convenient graphical form for a wide range of wing plan forms. These results do not, however, show an entirely satisfactory agreement with experimental data, the lift effectiveness being, in general, underestimated by about 10 per cent.

There is now a considerable amount of suitable experimental data covering a wide range of plan form, and thus it appeared that it might be possible to derive an empirical or semi-empirical method for estimating control lift.

ANALYSIS OF EXPERIMENTAL AND THEORETICAL DATA

The analysis of the experimental and theoretical data is dealt with in two stages, the data on full span controls being followed by that on part span effects, and since a large proportion of the experimental data on full span controls is for plain controls, this type is considered first.

Due to the fact that nearly all the data is limited to simple straight tapered wings with streamwise tips, only this class of plan form is considered. All the data has been obtained from tests on wings alone and the effect of the presence of a body has not been considered.

In general, the analysis applies only to wings in which the ratio of control chord to wing chord does not vary spanwise. Reference is, however, made to cases of varying control-chord ratios.

Full Span Plain Controls

The experimental data on the lift increment due to full span plain controls was found to cover a wide range of wing plan forms. Ranges of parameters covered were, A from 1·6 to 6·2, $\Lambda_{\frac{1}{4}}$ from 0° to 52° and λ from 0 to 1·0. Thickness-chord ratios for the wings varied from 6 to 12 per cent. and control-chord ratios from about 0·15 to 0·40.

To check this data against existing Data Sheets, a_2 was estimated from Data Sheet Controls 01.01.03 (Issued July, 1948) on the assumption that the ratio $(a_2)_0/(a_1)_0$ given by that sheet applied also to three-dimensional conditions, *i.e.* assuming that $a_2/a_1=(a_2)_0/(a_1)_0$. Thus a_2 was determined by multiplying $(a_2)_0/(a_1)_0$ by the value of a_1 found from Data Sheet Wings 01.01.01 (Issued March 1953). Values of a_2 calculated in this way were found to be about 10 per cent. below the corresponding experimental values, the error generally being greatest for wings of low aspect ratio, and it was thus apparent that the two-dimensional data given by Controls 01.01.03 was not directly applicable to the wings in question.

On the basis of the Multhopp method this error might be predicted since calculations using this method show that the ratio a_2/a_1 is generally slightly greater than the corresponding ratio $(a_2)_0/(a_1)_0$, the difference depending on wing plan form and on c_f/c. This then leads to a possible method of approach making use of theoretical values of

$$\frac{a_2/a_1}{(a_2)_0/(a_1)_0},$$

for if a_1, $(a_2)_0$ and $(a_1)_0$ are also known, a_2 may thus be determined.

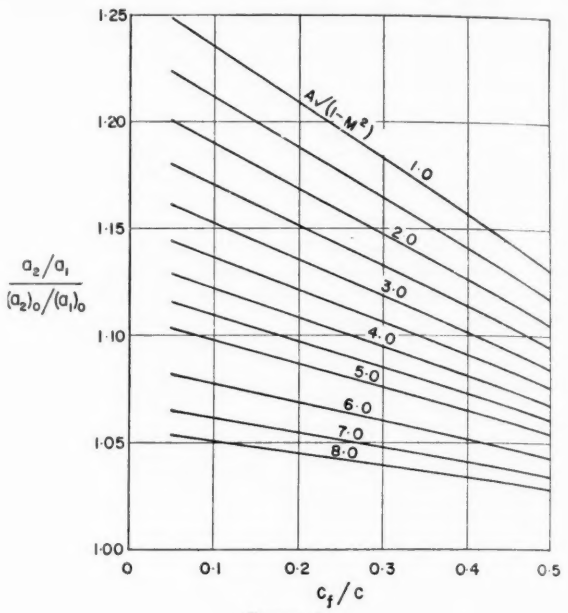

FIGURE 1.

Although the range of plan forms covered was not very great, calculations (made available by H. C. Garner of N.P.L.) indicated that the effect of wing taper ratio and sweep on the theoretical value of $(a_2/a_1)/[(a_2)_0/(a_1)_0]$ was small compared with the effect of wing aspect ratio and of c_f/c. It was expected that modification of these theoretical values would be necessary to allow for the effects of finite wing thickness, viscosity, etc., but the magnitude of this modification proved to be quite small. The experimental data confirmed that aspect ratio and control-chord ratio were the important parameters rather than sweep or taper ratio.

The resulting values for $(a_2/a_1)/[(a_2)_0/(a_1)_0]$ are shown in Fig. 1 in which the ratio is plotted against c_f/c for various values of $A\sqrt{(1-M^2)}$. The factor $\sqrt{(1-M^2)}$ is the usual compressibility correction factor.

Figure 2 shows, for the wings referred to above, the correlation between the experimental values of a_2 and the values determined using Fig. 1. For the determination of a_2 from Fig. 1, values of a_1 and $(a_1)_0$ were determined from Data Sheets Wings 01.01.01 and 05 respectively, and $(a_2)_0$ was found from the proposed revision to Data Sheet Controls 01.01.03. A similar correlation was obtained using the current issue of Controls 01.01.03 (July 1948), but the correlation from the revised sheet is shown here since the revised version will quite shortly be replacing the current issue.

[Since the issue of Controls 01.01.03 in 1948, more experimental evidence has become available and a revised version of the sheet has recently been approved by the Society's Aerodynamics Committee. This revision is based on work of H. C. Garner at N.P.L. and gives $(a_2)_0$ directly, rather than the ratio $(a_2)_0/(a_1)_0$. The general conclusions

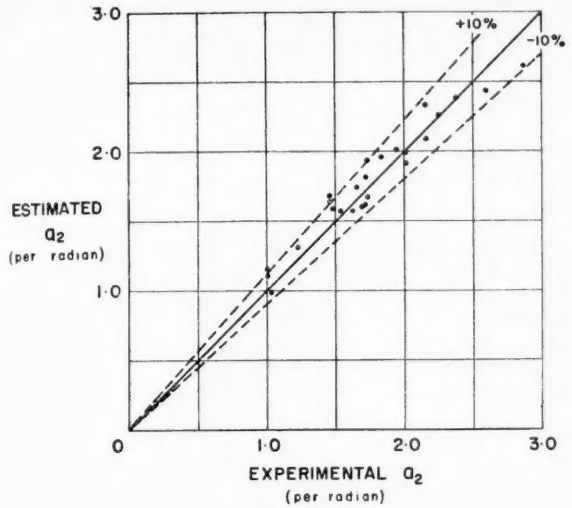

FIGURE 2.

of this note are unaffected by which version of Controls 01.01.03 is considered, but estimates in slightly closer agreement with experiment will result if the revised version is used in conjunction with Fig. 1.]

It can be seen from Fig. 2 that practically all the data is within ± 10 per cent. of the line of perfect correlation, and about half is within ± 5 per cent. Considering that any differences in this correlation are the summation of differences between Data Sheet calculations and experimental values for $(a_1)_0$, a_1, $(a_2)_0$ and $(a_2/a_1)/[(a_2)_0/(a_1)_0]$, this correlation could be considered as giving good support to the curves of Fig. 1. A new data sheet based on these curves has recently been approved by the Society's Aerodynamics Committee.

Where the experimental data was on wings with a control gap, the effect of the gap has been allowed for in the calculations by the method of Controls 01.01.04.

Full Span Split Flaps

The available experimental evidence on the lift increment due to full span split flaps is much more limited than that for plain controls and covers a considerably narrower range of wing plan forms. An analysis of the form carried out for plain controls, therefore, is not possible. Available data, however, was compared with values of lift increment estimated from Data Sheets in the following manner.

Using the appropriate sheet in the series Flaps 01.01.01 to 06, the lift increment may be determined for values of flap-chord ratio, wing thickness-chord ratio and flap deflection corresponding to the wing tested experimentally. This estimated value of lift increment would be that for a rectangular wing of aspect ratio 6, and can be denoted by $(\Delta C_L)_6$. From the curves of Fig. 1, the ratio $(a_2/a_1)/[(a_2)_0/(a_1)_0]$ can be determined both for the rectangular wing of aspect ratio 6, and for the wing tested. If these ratios are denoted by R and R_6 respectively, then

$$\frac{R}{R_6} = \frac{a_2/a_1}{(a_2)_6/(a_1)_6}$$

and thus

$$\Delta C_L = (\Delta C_L)_6 \times \frac{R}{R_6} \times \frac{a_1}{(a_1)_6}.$$

$a_1/(a_1)_6$ may be determined from Data Sheet Wings 01.01.01, and thus ΔC_L for the wing may be found.

Comparison of estimated and experimental values in this case showed considerably greater scatter than for the full span plain controls, covering a band of about ± 20 per cent., but the points were evenly distributed on either side of the line of perfect correlation.

This suggests, therefore, that, in the absence of a wider range of data and a more extensive investigation, the curves presented in Fig. 1 may also be applied to full span split flaps although the results will probably not be as accurate as for plain controls.

Full Span Slotted and Double Flaps

Insufficient experimental data has been found for any significant comparison of estimated and experimental lift increments due to slotted or double flaps. In the absence of other information, however, it is suggested that the lift increment be determined from the values for rectangular wings of aspect ratio 6 (*see* Data Sheets Flaps 01.01.08 and 09) in the manner outlined for split flaps.

Part Span Controls

For inboard controls, the variation of lift coefficient increment with control span is of the form shown by the curve in Fig. 3; the value of $(\Delta C_L)_{\text{full span}}$ depending on wing plan form, control-chord ratio, and so on. For controls other than inboard, it is usually assumed that the lift increment is the difference between the lift increments for two inboard flaps whose spans (b_{fo} and b_{fi}) are the distances between the outer and inner ends of the actual flap (*see* Fig. 3). If the maximum ordinate of the diagram of ΔC_L against η is now reduced to unity, the resultant curve will give a conversion factor to convert the full span value of ΔC_L (or a_2) to the corresponding part span value. In this way theoretical curves for part span conversion factors can be used in conjunction with empirical or semi-empirical data on full span controls.

The existing Data Sheet on part span effects (Flaps 01.01.07, issued June, 1944) is based on this approach, but

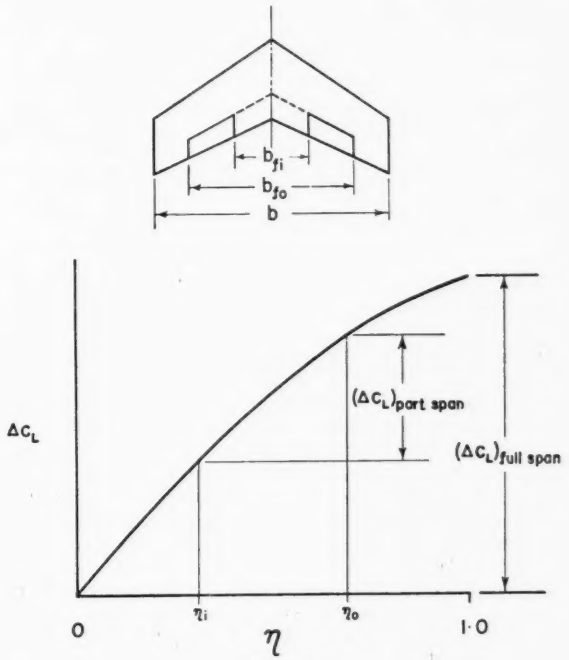

FIGURE 3.

was derived for unswept wings of aspect ratio 6. Comparison with experimental data shows that this sheet is not very reliable for swept wings or wings of low aspect ratio, and thus some revision is indicated.

The theoretical results of both Multhopp[2] and DeYoung[3] can be used to determined part span conversion factors. The available calculations by the former method, however, are not sufficient to cover a comprehensive range of wing plan form.

The Weissinger type calculations given by DeYoung do give part span conversion factors for a wide range of plan form, but direct presentation of these results would involve a large number of curves. While these curves give a fairly reliable estimate for part span effects, the scatter of the experimental data, even for results on similar wings, is quite large and thus some simplification of the presentation would seem to be justifiable.

Such a simplification is shown in Fig. 4. From this figure, part span conversion factors are found as follows:

If ΔC_L is the lift coefficient increment for the wing due to a full span control, and $(\Delta C_L)_{f_i}$ is the corresponding lift coefficient increment due to an inboard part span control of span b_f,

then $$(\Delta C_L)_{f_i} = K \times \Delta C_L$$

where $$K = k_1 [1 + k_2 (A - 6) + k_3 \sin \Lambda_{\frac{1}{4}}]$$

and k_1, k_2 and k_3 are factors given in Fig. 4. For controls other than inboard, these curves are used in the manner described at the beginning of this section.

It is seen that for $A = 6$ and $\Lambda_{\frac{1}{4}} = 0$, $K = k_1$ and thus the lower diagram gives directly the part span conversion factor for this type of plan form. The curves of this lower diagram have been only very slightly modified from the corresponding results given by DeYoung in order to improve the overall consistency. For unswept wings of aspect ratios other than 6, the part span effects obtained from Fig. 4 are within 2 per cent. of the results given by DeYoung. For swept wings the corresponding differences generally do not exceed 5 per cent.

DeYoung's results cover the range in plan form given by $A = 2$ to 10, $\Lambda_{\frac{1}{4}} = 0°$ to 60° and $\lambda = 0$ to 1·0, and the curves of Fig. 4 and the equation for K are derived to be applicable over the same range. For the lower limit to A, however, comparison with experimental data shows that Fig. 4 gives reliable results down to values of aspect ratio of about 1·5.

To allow for the effects of compressibility the subsonic linearised theory similarity law may be applied. Thus A is replaced by $A\sqrt{(1 - M^2)}$ and $\Lambda_{\frac{1}{4}}$ by

$$\tan^{-1}[(\tan \Lambda_{\frac{1}{4}})/\sqrt{(1 - M^2)}].$$

A comparison of experimental data with results estimated from Fig. 4 is shown in Fig. 5 in which experimental values of $(\Delta C_L)_{\text{part span}}/(\Delta C_L)_{\text{full span}}$ are plotted against K. This figure includes results from wings with plain controls, split flaps and slotted flaps. Although the discrepancies on a percentage basis are sometimes quite large, a detailed study of some of the data shows an appreciable amount of experimental scatter and a consistently reliable prediction of part span lift increment is therefore not possible. Thus it is considered that within the limits of experimental accuracy, Fig. 4 may be used for estimating the part span lift increment for wings with any conventional form of control or flap.

Such Multhopp calculations as are available on the lift coefficient increments for part span controls have been compared with results obtained from Fig. 4 and the agreement was in all cases within ± 5 per cent.

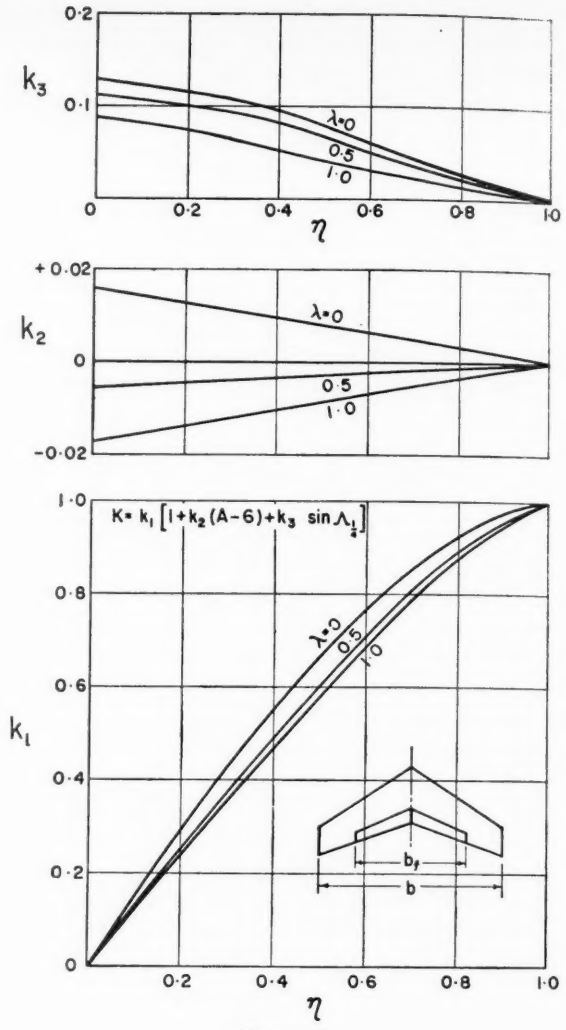

FIGURE 4.

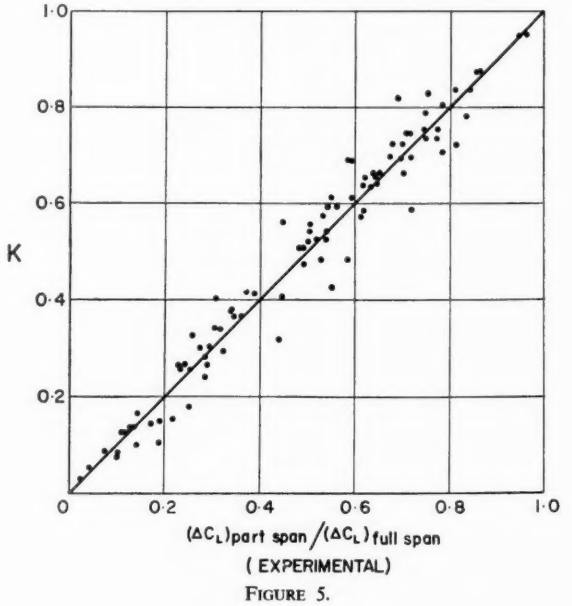

FIGURE 5.

Controls of Varying Control-Chord Ratio

There is insufficient experimental evidence for an analysis of the effect of a spanwise variation of c_f/c. In the absence of other information, therefore, such cases may be treated by a strip method. The total lift increment is then the sum of the separate lift increments for a series of part span sections of assumed constant c_f/c determined using Fig. 4.

CONCLUSIONS

The analysis of the available data has shown that the lift increment due to control deflection can be estimated to an acceptable degree of accuracy from the Royal Aeronautical Society's Aerodynamics Data Sheets, provided that certain additions and revisions are made. Such additions and revisions, based on Figs. 1 and 4 of this note, have been approved by the Society's Aerodynamics Committee and will shortly be issued to Data Sheet subscribers.

ACKNOWLEDGMENTS

The authors wish to acknowledge the advice given during the course of this work by the Society's Aerodynamics Committee and the assistance of Mrs. P. E. Measures in the computational work.

REFERENCES

1. Royal Aeronautical Society. Aerodynamics Data Sheets.
2. MULTHOPP, H. (1955). Methods for Calculating the Lift Distribution on Wings (Subsonic Lifting-Surface Theory). A.R.C. R. & M. No. 2884, 1955.
3. DEYOUNG, J. (1952). Theoretical Symmetric Span Loading Due to Flap Deflection for Wings of Arbitrary Plan Form at Subsonic Speeds. N.A.C.A. Report 1071, 1952.

Helicopter Stability in Hovering Flight

(Correspondence on the Technical Note by P. R. Payne in the September 1955 Journal, pp. 635-640)

A. R. S. BRAMWELL, Grad.R.Ae.S.

IN VIEW of the strong claims for offset flapping and drag hinges made by Mr. Payne in his Technical Note (September 1955) I would like to make the following comments.

1. Many of his conclusions depend on the assertion that there is a steady inertia force on the rotor hub when the rotor tilts. This force is simply the reaction on the hub due to the sideways thrust component and should not be included with the external forces on the helicopter. Neglecting the drag forces we have to satisfy the equation of the in-plane motion of the blade

$$\ddot{\zeta}\cos\beta - 2(\omega+\dot{\zeta})\dot{\beta}\sin\beta = 0 \quad . \quad . \quad (1)$$

If we assume that the lift variation is such as to produce flapping of the form

$$\beta = a_0 - a_1\cos\psi - b_1\sin\psi \quad . \quad . \quad (2)$$

i.e. coning plus sideways and backwards components of tilt, then substitution of (2) in (1) gives the corresponding motion in ζ. We find on resolving along the tip-path plane that the in-plane acceleration vanishes (in any case $\dot{\beta}$ is zero with respect to the tip-path plane) and that the component in any other direction has the corresponding component of thrust to produce it.

2. It appears from Mr. Payne's equations that the difference between rotors having drag hinges and those without depends on the term $\left(1+\frac{d^2\zeta}{dt^2}\right)$. It is difficult to see the significance of this term as its value surely depends on the azimuth position of the blade whereas the expression containing it is the *mean value* of pitching moment in the fore and aft direction. Besides, the dimensions of the terms in the bracket differ as they do also in equation (5) of the note.

3. Since there is no difference in stability between helicopters having drag-hinges and those without, it is untrue to say the Bell and Hiller helicopters need stabilising bars because the blades lack drag hinges. The stabilising bar having been invented, the two-bladed see-saw rotor was the simplest system to which it could be applied.

4. In deriving expressions for $\partial a_1/\partial V$ and $\partial a_1/\partial\Omega$, Mr. Payne has assumed that b_1 is small. If we use his equation from Ref. 6 of his Note and solve them exactly we find that b_1 is by no means small but in some cases may even be larger than a_1. The values of M_V and M_Ω calculated for the Sikorsky S-51 for different blade inertias are

ξ	0	0·1	0·2	0·3	0·4	
M_V	23·8	50·8	90·2	110	122	$\gamma_0=0{\cdot}4$
M_Ω	−752	−2,380	−4,010	−5,790	−6,600	
M_V	23·8	65·0	123	135	132	$\gamma_0=2/3$
M_Ω	−1,273	−5,430	−9,500	−12,550	−12,700	
M_V	23·8	98·8	166	143	112	$\gamma_0=4/3$
M_Ω	−2,530	−18,100	−27,100	−28,100	−22,700	
M_V	23·8	164	148	72·2	46·3	$\gamma_0=4$
M_Ω	−7,600	−95,000	−76,100	−46,000	−29,400	

where γ_0 is the inertia number of a uniform blade when $\xi=0$.

These values give the following period of oscillation (P secs.) and time to double amplitude (T_D secs.)

ξ	0	0·1	0·2	0·3	0·4	
P	15·8	12·5	10·6	10·2	10·1	
P (Hohenemser)	6·2	7·6	7·3	8·0	8·1	$\gamma_0=0{\cdot}4$
T_D	3·6	3·4	3·4	3·8	4·3	
P	15·8	12·5	11·0	11·3	11·6	
P (Hohenemser)	8·1	10·1	9·4	10·7	10·8	$\gamma_0=2/3$
T_D	4·0	5·0	6·0	7·5	8·3	
P	16·4	15·2	14·5	15·2	15·6	
P (Hohenemser)	11·4	15·0	14·1	15·5	15·7	$\gamma=4/3$
T_D	5·3	19·6	25·6	30·2	25·6	
P	20·4	27·5	25·0	27·3	27·5	
P (Hohenemser)	19·8	26·6	25·0	27·2	25·2	$\gamma=4$
T_D	15·8	333	216	166	104	

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The results show that apart from ξ the stability depends greatly on blade inertia. (The standard S-51 blades are represented by $\gamma_0=2/3$.) Thus, for light blades, offset hinges have little effect on phugoid damping but cause a considerable fall of period, while for very heavy blades the phugoid is almost neutrally stable above $\xi=0{\cdot}1$ with a longer and almost constant period. It should be pointed out that for the larger values of ξ and γ_0, b_1 is also too large to neglect and indicates strong coupling between lateral and longitudinal motions. All the above figures, however, have been calculated on the assumption that the longitudinal and lateral motions can be treated separately and therefore may be somewhat in error for large ξ and γ_0.

Incidentally, the sign of b_1 in the equation (6) of Mr. Payne's note is incorrect and therefore the expressions for a_1 and $\partial a_1/\partial\Omega$ in equations (8) and (13*a*) should be prefixed by a negative sign. Hohenemser's expression for the period has been misquoted to allow for the errors of sign. The above figures show that Hohenemser's expression gives very poor agreement at the lower values of ξ and γ_0.

5. Although Mr. Payne's article was concerned only with hovering flight, presumably for simplicity, the use of offset hinges raises two important considerations relating to forward flight.

(*a*) The effect of offset hinges is to increase the moment due to rotor tilt and the above figures show that M_V and M_Ω are increased in roughly the same proportion. With forward speed, however, we must consider the moment due to incidence (in usual stability notation, the term M_w) and this term, too, can be expected to increase in proportion to M_V and M_Ω. But it is well known that M_w, which is *positive* for the single-rotor helicopter, is responsible for its poor handling qualities at the higher forward speeds, so that increasing its numerical value by offset hinges may well make the handling qualities unacceptable.

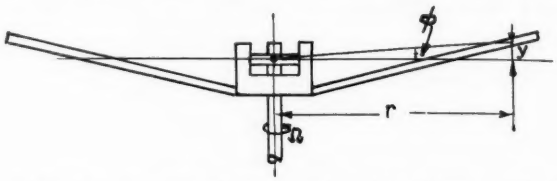
(*b*) It has been suggested that since for most speeds the rotor thrust increases with increases of forward speed and incidence, then setting the helicopter c.g. forward of the rotor force axis will provide stabilising moments with respect to forward speed and incidence. (This will not be true for the hovering case or for any forward speed so low that an oscillation would cause the machine to travel *backwards* during the motion.) The disadvantage of this suggestion is that a steady pitching moment from the rotor hub must be supplied to trim the helicopter which would be oscillatory with respect to the hub (which rotates) and would no doubt impose severe fatigue problems. In any case, there is sure to be *some* fuselage pitching moment which requires trimming thus giving rise to an oscillatory hub moment.

To sum up, it appears that a worthwhile improvement of stability can be achieved only when the blades are very heavy but that in all cases the handling characteristics, which are already poor, would be made even worse. It seems unlikely that offset hinges could compete with an automatic pilot or even a stabilising bar which, after all, is a very simple device. A stabilising bar, well designed, should provide positive stability—which offset hinges could never do—without adverse effects on the controllability. Nevertheless, Mr. Payne is right in suggesting that means should be explored before we commit ourselves to the complication of automatic pilots.

LEONARD GOLAND

(*Chief Project Engineer, Forrestal Research Center, U.S.A.*)

IN THE SECTION entitled "Damping in Pitch" (page 637) it is shown that for the case of rotor having no drag hinges, the damping in pitch is considerably smaller than a similar one with drag hinges, if the flapping pin offset is small. I feel, however, that the results are based on an additional assumption of practical importance. This assumption is that the rotor is not underslung (*see* figure). Most existing rotors of the type mentioned (Bell and Hiller) are underslung so that the c.g. of the blade system is in the plane of the feathering and flapping axes. This being the case, the in-plane Coriolis forces due to flapping do not give rise to a constant force and therefore do not reduce the conventional "damping in pitch" term. Consequently the damping in pitch for a Bell or Hiller type



is similar to that of rotors with drag hinges. Blade flexibility would naturally modify these statements. This may also be explained mathematically as follows:— The Coriolis acceleration may be stated as:

$$\bar{a}_c=2\bar{\Omega}\times\bar{V}$$

where V denotes linear velocity.

The in-plane component normal to the blade longitudinal axis acting on a blade mass particle is therefore,

$$a_{CI}=2\Omega\left(\frac{r}{\cos\phi}\dot{\phi}\sin\phi\right).$$

Integrating along the underslung blade we have therefore,

$$a_{CI}=2\Omega\dot{\phi}\int_{\text{blade}} r\tan\phi\,dm=2\Omega\dot{\phi}\int_{\text{blade}} y\,dm=0.$$

However, there is created, by the in-plane Coriolis accelerations, a moment about the cyclic feathering axis of magnitude $2\Omega\dot{\phi}\int_{\text{blade}} y^2dm$ which contributes to blade "weaving."

In reality, the Bell and Hiller rotor hubs are underslung, so that the centres of gravity of the blades lie on the feathering axis, to prevent the occurrence of the forces which Mr. Payne considers. Therefore, I cannot accept the statement: "This (aforementioned reduction in pitch damping due to Coriolis forces) may explain why all successful helicopters whose rotors have no drag hinges and zero flapping hinge offset possess some form of automatic stabilising device, such as Bell or Hiller systems."

Received 24th October 1955

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IN MY OPINION the analysis in Mr. Payne's Technical Note is in error as far as the effect of drag hinges on the rotor derivatives is concerned. This error was originally made in Mr. Payne's article in the June 1955 issue of "Aircraft Engineering" and was introduced from there into equation (9) of the reference note.

Received 23rd November 1955

Upon the assumption of a uniform angular velocity of the rotor shaft, Mr. Payne arrives at the conclusion that depending on whether or not the blades are articulated by drag hinges there are large differences in tangential rotor force. According to Mr. Payne's analysis these differences are caused by Coriolis forces which accompany the flapping motions of a coned rotor blade in the absence of drag hinges. By changing from a rotor with drag hinges to one without drag hinges very little change in the aerodynamic forces on the rotor is found, and Mr. Payne was justified in neglecting these changes. If, however, the aerodynamic loads on the rotor do not change the resultant rotor force must also be unchanged, and this includes the tangential force. Internal inertia forces like the Coriolis forces can never produce an external force on the aircraft, this can only be done by air loads.

The reason why Mr. Payne erroneously assumed that the Coriolis forces have the character of an external force must be found in his erroneous assumption of a uniform angular velocity of the rotor shaft. The angular velocity of the rotor shaft has to be determined according to Euler's equations and will be non-uniform in such a way as to compensate for the effect of the Coriolis forces.

The " Coriolis " term has to be omitted in all equations of the note and the conclusions based on the effect of this term have to be revised. It is not true that rotors without drag hinges like the Bell and Hiller rotors have a lower damping in pitch because of the absence of drag hinges and, therefore, need artificial damping devices. It is also not true that see-saw rotors with tip mounted engines, because of the absence of drag hinges, would be very unstable and difficult to fly without stabilising devices. The presence of the drag hinge *does not* improve helicopter stability.

Reply by P. R. PAYNE

THE COMMENTS which have been made in reply to my original Technical Note are of considerable interest, particularly those of Mr. Bramwell. The criticism common to all of them relates to the effect of drag hinges on helicopter stability, which all three correctly aver to be negligible. This was first pointed out by V. A. B. Rogers[1] and the proof was demonstrated by H. L. Price[2] using the principles of d'Alembert. It is interesting to note that although the present writers agree with Ref. (1) and (2), they all advance different and apparently erroneous reasons. The correct explanation of the error is that a small angle assumption ($\cos\beta = 1{\cdot}0$) was made in Ref. (4) which prevents the radial accelerations on the blade from appearing in the analysis.

Mr. Hohenemser ignores the radial accelerations, and claims that the steady force does not exist because the angular velocity of the shaft is not constant when no drag hinges are fitted. A little thought will show that first harmonic normal accelerations alone cannot cause the shaft to oscillate, because they vary in a sinusoidal manner: if the shaft did in fact oscillate it would certainly complicate mechanical design, particularly of such transmission items as gears and free-wheels.

Mr. Goland's " additional assumption " is also not the reason for the error, as we have seen above, and the under-slinging of the rotor only affects the fluctuating component of the resulting forces at the hub. Thus two-bladed rotors are under-slung in order to reduce Coriolis vibration.

Mr. Bramwell's argument seems even more oblique, since he has quoted equations relating to a rotor with " ideal " (centrally-located) drag hinges to discuss one which has no drag hinges. The " Coriolis force " is not balanced by a thrust component, but never did exist to be balanced by anything.

His next two points also relate to the force which is agreed to be mythical, but with the third the author unreservedly agrees. Indeed, as Mr. Bramwell is aware, the author made this point last November in Ref. 6, and gave numerical examples to emphasise it. Without question the traditional assumption that " b_1 is small " must be discarded for rotors with flapping constraint.

Mr. Bramwell's main point is that the use of Hohenemser's approximation for the period of oscillation is not justified for helicopters with light blades, and that in such cases an increase in flapping pin offset may lead to a deterioration in stability characteristics. This is undoubtedly true for the S.51 example which he quotes, and this reservation should have been made in the original Technical Note. It was not because the writer has not personally been concerned with helicopters whose stability characteristics are as poor as this. As the S.51 figures show, Hohenemser's approximation is satisfactory provided that the stability is reasonably good, so that its use in a Note which discusses a method of obtaining further improvements seems to be acceptable.

The improvement in stability with flapping constraint is amply justified in general by other references.

For example, R. H. Miller has given the following results in his classic paper[5] on hovering stability.

Offset parameter $\xi = 0{\cdot}033$	0·33	Rigid rotor
Period P 9 secs.	11 secs.	11 secs.
Time to double amplitude T_D 3 secs.	21 secs.	22 secs.

In the ultra-light class of helicopter, with very heavy (ram-jet driven) blades, the example of Ref. 6 gives an even more dramatic improvement of stability with hinge restraint. Noting that stiff and offset hinges are exactly analogous we have:—

$(\chi\xi) =$	0	0·5	1·0	4·0
P	26·45	26·8	27·0	28·9 secs.
T_D	23·25	140·5	190·0	117·3 secs.

It is important to develop a theory which embraces and explains the apparent contradiction of Mr. Bramwell's contribution, and this can be done by studying the cubic equation which defines the motion of a hovering helicopter. This approach also justifies Hohenemser's approximation for period P, and enables rapid answers to be obtained even in the regions where Hohenemser's approximation does not apply. The characteristic equation is

$$\lambda^3 + a\lambda^2 + b = 0.$$

where $$a = \frac{M_\Omega}{I} + g\frac{\partial a_1}{\partial V}$$ (the second term being small)

$$b = gM_V/I$$

where M_Ω and M_V are given in the original Technical Note, and I is the helicopter moment of inertia about the c.g. The roots of this cubic are generally one real and negative root, and two complex roots, which may be written as $(m + in)$ and $(m - in)$. Substituting the complex forms into the equation, expressions can be obtained for

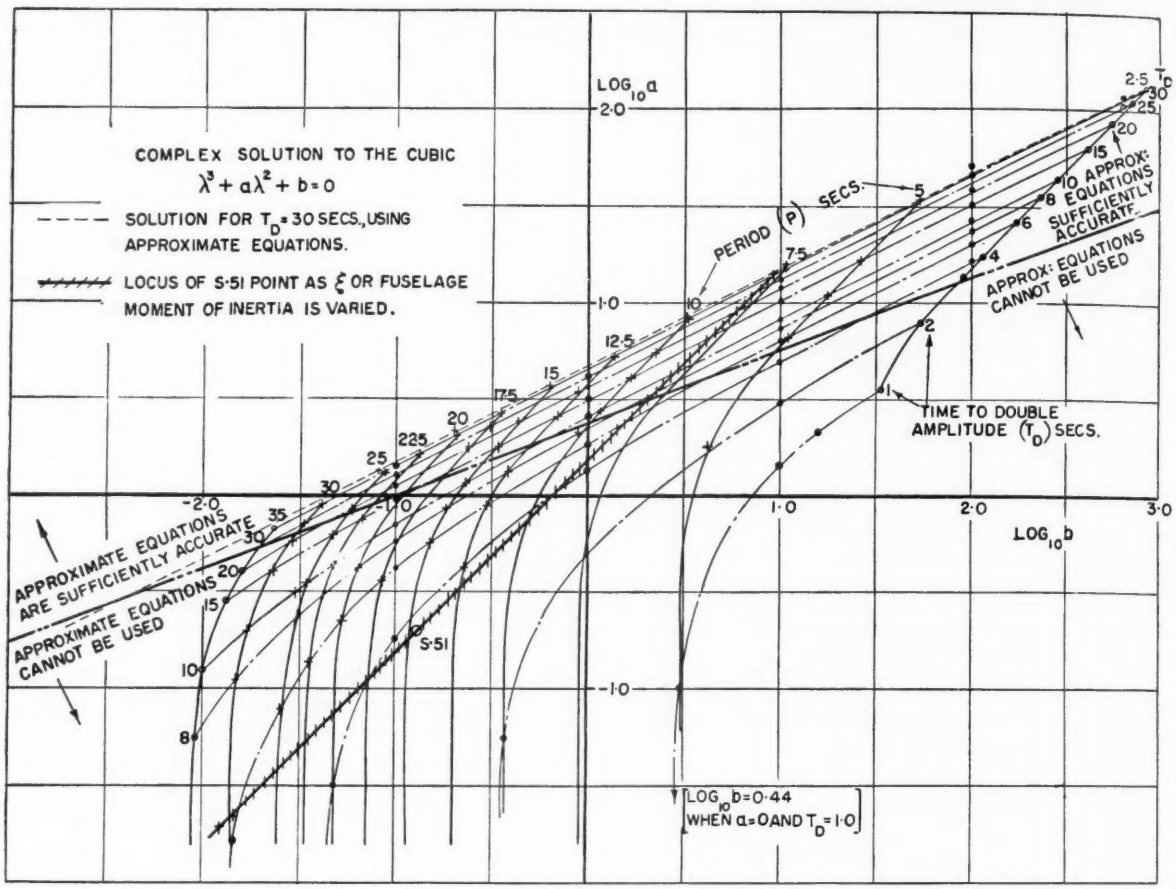
COMPLEX SOLUTION TO THE CUBIC
$\lambda^3 + a\lambda^2 + b = 0$
SOLUTION FOR T_D = 30 SECS., USING APPROXIMATE EQUATIONS.
LOCUS OF S·51 POINT AS ξ OR FUSELAGE MOMENT OF INERTIA IS VARIED.
$\log_{10} a$
$\log_{10} b$
PERIOD (P) SECS.
TIME TO DOUBLE AMPLITUDE (T_D) SECS.
APPROX: EQUATIONS CANNOT BE USED
10 APPROX: EQUATIONS SUFFICIENTLY ACCURATE
APPROXIMATE EQUATIONS ARE SUFFICIENTLY ACCURATE
APPROXIMATE EQUATIONS CANNOT BE USED
S·51
$\log_{10} b = 0·44$ WHEN $a = 0$ AND $T_D = 1·0$

FIGURE 1.

a and b in terms of m and n

$$a = \frac{n^2}{2m} - \frac{3}{2}m$$

$$b = mn^2 + \tfrac{1}{2}\left(\frac{n^4}{m} + m^3\right).$$

If $3m/2$ is small compared with $n^2/2m$
mn^2 is small compared with $n^4/2m$
m^3 is small compared with n^4/m

then the following approximations may be used for period of the phugoid (P) and the time to double amplitude (T_D)

$$P = \frac{2\pi}{n} = 2\pi\sqrt{\left(\frac{a}{b}\right)} \text{ (Hohenemser's approximation)}$$

$$T_D = \frac{\log_e 2}{m} = \frac{2a^2}{b}\log_e 2.$$

Replacing the coefficients a and b by the damping derivatives

$$P = \frac{2\pi}{\sqrt{g}}\sqrt{\left(\frac{M_\Omega}{M_V}\right)}$$

$$T_D = \frac{2}{g}\left(\frac{M_\Omega}{M_V}\right)\frac{M_\Omega}{I}\log_e 2.$$

When the limiting approximations are *not* valid, no simple analytical result can be obtained, but the equations for a and b can be used to plot a carpet of solutions. This has been done in Fig. 1 to cover all the likely values of P and T_D below $P = 35$ secs., and $T_D = 30$ secs. The region where the approximate equations apply is also shown in Fig. 1, and in the author's opinion this should cover most future designs, for the very cogent reason that if they are not in this region their stability can hardly be considered satisfactory.

Some earlier calculations by Mr. Bramwell for the S.51 ($\xi = 0$) are shown in Fig. 1. Increases in ξ or variations in fuselage moment of inertia I will cause this point to move across the paper at 45° to the axes. Increasing ξ will move the point up and to the right, asymptoting to a period of 8 secs. (as Mr. Bramwell avers in his earlier work), while T_D increases only very slowly for the low values of ξ, the increases becoming larger with higher values of ξ. Reducing the fuselage moment of inertia has exactly the same effect; in the case of this example the graph shows that the stability characteristics will be improved by an *increase* of the fuselage moment of inertia, which flatly contradicts many published generalisations!

A different example helicopter would enable different conclusions to be reached, depending on its position in Fig. 1. Thus the author accepts Mr. Bramwell's criticism to the extent of adding a qualification to the original remarks. Hinge offset or elastic constraint is beneficial, increasing the time to double amplitude but not affecting

the period, *provided that* the values of P and T_D are already fairly good for $\xi=0$. This reservation could be read into the original Note in so far as it was concerned with the enlightened designs of the future rather than the veterans of the past. Moreover it pre-supposes the suggestion already made, that greater stability should be obtained by aerodynamic design rather than the manifold complications of "black boxes."

For stability in forward flight Mr. Bramwell's suggestion of placing the c.g. forward of the shaft (previously made by F. H. Robertson[7]) is an excellent example of this technique. The large hub moments would seem to be unimportant, since hub moments are essential if the benefits of hinge constraint are to be realised, even in steady flight. To take the simplest example, hub moments would present no design problem at all with a stiff-hinged see-saw rotor, for which the author believes there is a big future.

It is difficult to accept that M_u (the change of pitching moment with angle of attack, due to the rotor) will cause the deterioration in forward flight stability which Mr. Bramwell expects with constrained flapping. For two degrees of freedom in forward flight the characteristic equation is of the form

$$a\lambda^3+b\lambda^2+c\lambda+d=0.$$

The term M_u appears in the equation for c and d in such a manner that it can be balanced by both c.g. position and the rate of change of the fuselage pitching moment with α, with the exception of one term in the static stability coefficient d: here it can be balanced by the derivative of fuselage pitching moment with respect to the forward speed. Mr. Bramwell can only have reached his conclusions by neglecting fuselage pitching moment and c.g. position in his analysis, since the more complex results for three degrees of freedom lead to the same general conclusions.

If the degrees of freedom are limited, the theory of helicopter stability in forward flight is mathematically simple but unfortunately includes many terms. The present type of stability "investigation" which explores the effect of small variations in one or more variables in the vicinity of an existing design would seem to be useless, in that it can lead to highly misleading conclusions about general trends, of the type mentioned above for the much simpler case of hovering flight. It would be a very great help to helicopter designers if "someone" could be persuaded to set up the equations on a computor and explore exhaustively the effect of all the variables, taking them well beyond the range of values at present considered reasonable. It should be possible to present the results of such an investigation in a way which would be extremely valuable to design teams, but since it would not appear to fall within the province of A and A.E.E., it is difficult to see who would undertake such a task.

REFERENCES

1. ROGERS, V. A. B. (1955). Letter to the Editor *Aircraft Engineering*. Vol. XXVII, No. 319, September 1955.
2. PRICE, H. L. (1955). Letter to the Editor *Aircraft Engineering* Vol. XXVII No. 321 November 1955.
3. PAYNE, P. R. (1955). The Control of Helicopter Rotor Vibration. Lecture read before the Helicopter Association of Great Britain on October 7th 1955.
4. PAYNE, P. R. (1955). Helicopter Rotor Vibration in the Tip-path-plane. *Aircraft Engineering*, Vol. XXVII, No. 316, June 1955.
5. MILLER, R. H. (1948). Helicopter Control and Stability in Hovering Flight. *Journal of the Aeronautical Sciences*, August 1948.
6. PAYNE, P. R. (1955). The Stiff-hinged Helicopter Rotor. *Aircraft Engineering*, Vol. XXVII No. 321 November 1955.
7. ROBERTSON, F. H. (1955). Letter to the Editor *Flight*, p. 849, December 2nd 1955.

Power Loss and Fuel Pressure Rise Due to Centrifugal Pumping in Tip-Driven Helicopter Rotors

by

A. CZERNIN

AS one being intimately concerned with the development of the jet-propelled helicopter for the past thirteen years, I would like to comment on some of the statements in the Technical Note by P. R. Payne, published in the May 1955 JOURNAL, which I feel are misleading and indeed wrong.

NOTATION

ρ	density of fuel
Q	torque about hub centre
V_{rad}	radial velocity of fuel in blade
ω	rotational speed of rotor
R	radius of rotor
V_j	speed of jets in full expansion nozzle
V_T	tip speed
M	air mass flow per second
F	fuel mass flow per second
T_h	tip engine thrust
A	connectional area of fuel pipe

Received 10th December 1955

The author claims that by using small bore fuel pipes and thereby high fuel velocities, pressures higher than $\frac{1}{2}\rho V_T^2$ can be achieved (Fig. 2) at the blade tip and this without taking recourse to pressurising the fuel before it enters the rotor. It is difficult to see how, with an applied torque of $Q=\rho A V_{rad}\omega R^2$, pressures higher than $\frac{1}{2}\rho V_T^2$ can be possible as, neglecting friction losses, the spray velocity equivalent to this pressure can be shown to be V_T. The applied torque is therefore equal to the torque produced by the rearward facing fuel spray: $Q=\rho A V_{rad}\omega R^2=FV_TR$. Any pressure higher than $\frac{1}{2}\rho V_T^2$ would mean that the energy content of the fuel at the tip is higher than the energy invested by the applied torque—a condition which would take us a long way towards the desirable goal of perpetual motion.

In fact, this is of course impossible from purely physical considerations. Mathematically, too, it is seen to be impossible once the assumption that $\cos\phi\simeq 1{\cdot}0$ (before equation 8) is dropped, as it must be, $\cos\phi$ being θ at the blade root and approaching 1 at the tip.

Discarding Mr. Payne's incorrect assumptions, the examination of pressure changes through a fuel system with *metering at the tip* (*i.e.* fully filled fuel pipes) becomes again the simple physical problem it really is. The total pressure rise is simply $\frac{1}{2}\rho V_T^2$ with a deduction for frictional pressure drop. The fuel pipe diameter can influence only this frictional pressure drop.

Metering the fuel prior to its entering the rotating blades, a method widely if not exclusively, practised, leads to rather more complex conditions inside the rotating fuel lines. It would be interesting to hear Mr. Payne's views about this.

The author deals also with the power loss due to centrifugal pumping and spraying of the fuel, supposedly of the order of 6-12 per cent. of the rotor power in the case of ram-jets and even more with pressure jets.

The jet thrust of the pressure jet helicopter, neglecting its combustion chamber drag, is

$$T_h = M(V_j - V_T) + F(V_j - V_T).$$

Assuming stoichiometric air/fuel mixture ratio (15/1) and, as an average figure, $V_j \simeq 5V_T$ the term FV_T, *i.e.*, the power loss due to fuel pumping, becomes 1/65 of the rotor power or just above 1·5 per cent. for radial fuel spraying and just above 3 per cent. for upstream injection. The usual mixture ratios being normally rather 20/1 to 30/1 and the fuel injection radial, if not downstream, it becomes obvious that the fuel pumping power is of the order of 0·8 to 1·2 per cent. of the rotor power.

Due to the small ratio V_j/V_T of the low pressure jet, a higher percentage (not a lower one, as Mr. Payne states) will be needed for fuel pumping on ram-jet rotors.

However, $F(V_j - V_T)$ is always positive as $V_j - V_T > 0$, and the fuel mass contributes positively to the jet thrust. The use of an (imaginary) weightless fuel would reduce the rotor performance by an appreciable margin, though cutting fuel pumping work down to nil.

P. R. PAYNE

Mr. Czernin's correction does not appear to be in accordance with the facts, nor does it agree with current centrifugal compressor theory. The increase of pressure with radial velocity may easily be demonstrated in the following way, which is perhaps more obvious than the one I used in my original note.

Kinetic energy of the fuel $= \frac{1}{2}\rho q U^2$

Resultant velocity of the fuel at the rotor tip

$$U = \sqrt{(V_T^2 + V_{rad}^2)}$$

Therefore K.E. $= \frac{1}{2}\rho q (V_T^2 + V_{rad}^2)$

$$\text{Pressure} = \frac{\text{K.E.}}{\text{fuel flow}} = \tfrac{1}{2}\rho V_T^2 \left[1 + \left(\frac{V_{rad}}{V_T}\right)^2\right]$$

where the notation is the same as that of my original note. The actual increase due to V_{rad} is not as great as that obtained on the original assumption of $\cos\phi = 1{\cdot}0$, but it is still of the same order as the frictional loss in the pipe, and I imagine it might be possible to prove that the pressure rise to V_{rad} is always exactly equal to the frictional loss in a full pipe.

Mr. Czernin's equation for thrust relates to the gross thrust rather than the " stub " value, so that he naturally gets a lower percentage thrust loss due to pumping. To neglect drag is quite inadmissible for a ram-jet; hence the reason for my figure being as high as 6 per cent. In stating that the " loss " was even higher for pressure jets I was of course including the mass of air as well as the fuel.

On general considerations, taking Mr. Czernin's notation, the gross thrust of a ram-jet will be

$$T = MV_j + FV_j - MV_T - FV_T$$

the last term being the " pumping loss."

$$T = (M + F)(V_j - V_T).$$

This is Mr. Czernin's expression, and also the normal one for performance estimates. The important point about this equation is surely that it already allows for the " pumping loss " and also any other losses in the system. It is irrelevant whether the fuel pipe is full or half empty, or whether the fuel is sprayed forward or aft. This last point is easily checked on a static rig where it is found that operation of a forward facing spray causes a re-distribution of static pressures over the diffuser, giving an increase in diffuser " thrust " which exactly balances the " negative thrust " on the spray bar.

Further Comment by A. CZERNIN

I AM GLAD to see that Mr. Payne has now discarded his original formula for pressure rise in whirling fuel blades.

However, Mr. Payne still holds that higher pressure can be obtained at the tip (ignoring friction) with small diameter tubes than with large ones. This is still incorrect. All we can do by decreasing the pipe diameter is to increase the velocity head of the fluid, but at the expense of static pressure. The *total* pressure at any radial station remains constant irrespective of pipe diameter (neglecting friction) and conditions follow simply Bernoulli's law. The maximum pressure rise in such a whirling fuel pipe is $\frac{1}{2}\delta V_T^2$ and the resulting fuel speed through the orifice equals V_T. Higher fuel pressures can only be achieved by pressurising the fuel before it enters the rotor, but not by decreasing the fuel pipe diameter.

As to the effect of forward or backward facing fuel sprays, I cannot but repeat what I said in my first answer. Substituting now the ram-jet for the pressure jet, as Mr. Payne does, just veils the point in question by introducing the secondary effects like increased " diffuser thrust." This last is absent in the pressure jet, and a fuel spray in or against the direction of motion will leave the negative or positive effect as I mentioned.

The inclusion of the air flow while considering the " pump loss " of pressure jet rotors does indeed increase this loss to higher relative values than the pump loss due to fuel alone accounts for the ram-jet. As Mr. Payne at the beginning of his notes, referred to " the action of the fuel flowing along the blade " it was by no means self evident that this should include the air flow (approximately 20 times as much as the fuel flow) as well.

Further Reply by P. R. PAYNE

I AM afraid that I still cannot agree with Mr. Czernin's thesis about the pressure rise. As mentioned in my first reply to him, the situation is surely analogous to that of a centrifugal compressor with zero slip. In this case the static pressure rise is independent of air mass flow (corresponding to the rise of $\frac{1}{2}\rho V_T^2$ in fuel pressure) and the radial velocity, which increases as air mass flow and is induced by the impeller, represents a gain in kinetic energy. In the ideal case with zero slip and a 100 per cent. efficient diffuser, this kinetic energy can be converted into an additional static pressure rise of $\frac{1}{2}\rho V_T^2$ as well, because the radial velocity of the air at the impeller tip is equal to the circumferential velocity. If air mass flow is varied in a practical compressor, then for zero mass flow the static pressure at outlet is $\frac{1}{2}\rho V_T^2$; as the air mass flow increases (holding constant impeller r.p.m.) the static pressure rise in the diffuser rises, reaching a maximum value when extraneous effects such as compressibility nullify the beneficial effect of increased radial velocity.

My preoccupation with ram-jets was due to the fact that the note was originally concerned with this type of power unit. I am conscious of a beam in my own eye with regard to the looseness with which I defined the boundaries of the problems; neither the manner in which the fuel arrived at the blade root nor the upstream spray momentum boundary were defined. Such a boundary is most important when discussing the latter case. In the free flight case it is simple, since the fuel is travelling with the ram-jet, and the thrust is

$$T = M(V_j - V_T) + FV_j$$

This is the expression always used in ram-jet performance calculations, with modifications for supersonic flight. As mentioned in my original reply, we subtract the additional term FV_T when the fuel is initially stationary, this representing the change of momentum of the fuel between the hub and the ram-jet. The forces acting on the spray gear within the ram-jet are internal effects, and will not affect the change of momentum across the momentum boundary unless they modify V_j. Apart from possible aerothermopressor effects if the spray is injected in the diffuser, I cannot at the moment see how the spray direction can affect the total pressure loss through the unit. If this is true of the ram-jet, it must also be true for the pressure jet.

(Note: Correspondence on these two Notes by Mr. Payne is now closed.—Ed.)

Graduates' and Students' Section

ONE of the 1955-56 members of the Graduates' and Students' Section Committee, who recently paid a visit to firms in the Canadian aircraft industry, has written the following notes, in which he relates some personal observations on the ideas expressed on this page in the May and July issues of the Journal.

IN MAY, graduate and student members of the Society were urged to "aim self-critically at a high absolute standard, not blindly adjusting to the mediocre norm." By doing a job with speed, care, accuracy and the full application of intelligence, we were told, we would achieve more contentment, less frustration—and less tiredness. And then, in July; "there is probably a lack of appreciation by the young men of today of how much better off they are than their fathers were . . . the present-day engineer can show his gratitude for the facilities he enjoys by putting out a much greater effort for the hours he has to work".

But, graduates and students will have asked, is this the whole picture? Why not take a look at industry itself, for instance, and at the scope for speed, care, accuracy and intelligence—and the justification for gratitude?

In the March issue of the *Canadian Aeronautical Journal* appeared an article with the title "A Progress Report on the Canadian Aeronautical Engineer", in which the movements, status and opinions of about one hundred aeronautical engineers who graduated from the University of Toronto over a three-year period were analysed. As Toronto awards the only Canadian degree in aeronautical engineering, this analysis was most informative. It is not practicable to summarise the entire paper here, but one or two extracts are significant when considering the ideas quoted above.

In an average of 5½ years' experience since graduation, the "average" Canadian graduate in civil employment had changed his job once (more precisely, he had had 1·9 employers). Approximately one-third of those who replied to the questionnaires were thinking of changing their jobs within the next five years. Many of those who seemed content with their prospects of promotion in the Canadian industry, and most of those who did not, were unhappy with the attitude of their employers, and there was one particularly revealing comment by one graduate: "There was recently a great cry of shortage of engineers for aircraft work, but *efficient use of present staff is ignored*."

This, indeed, is a fundamental point for the British industry to consider. Where employers are *not* making full use of their young graduates' capabilities—which is "efficient use" for the firm as well as satisfaction for the individuals—then talk of "gratitude for the facilities they enjoy" is nonsense. And the "mediocre norm" is that imposed by the company. Graduates do not "blindly adjust" to this norm—the only blindness is that of the firm's management—instead, they are forced against their intelligence to fit into an inefficient, wasteful system. Some self-critical aiming at high standards, one might suggest, could usefully be done by industry, as well as by graduates and students.

This problem has arisen both in the United Kingdom and in Canada. And so the "news of members" columns in this Journal continue to record the movements of members from this country to Canada—and the corresponding columns of the *Canadian Aeronautical Journal* record movements of C.A.I. members from Canada to the U.S.A. Obviously one cannot generalise on the reasons for such moves; nevertheless (from the Canadian source already mentioned and from U.K. examples which the writer can recall) an enlightened company attitude has in many cases been a more important factor than the marked increase in salary. The grass still *seems* greener on the other side—and, in many cases, it certainly *is* greener.

Any industry which uses the brains of the majority of its graduate employees to conceive, design, produce, test and develop weapons of war has a special responsibility to use this talent with imagination and understanding. Yet these qualities of imagination and understanding are lacking in a wide section of the British Aircraft Industry, as many graduates and students of the past ten years or so are well aware. For it is the graduates and students who have met and who have recognised the restrictive aspects of traditional company attitudes. Too many of us have been tolerated as absolute nuisances for wishing to use our time well during vacation and graduate apprenticeships. Too many of us have noted rather fundamental differences between the scope of work offered in employment advertisements and at interviews, and the scope actually available after one has joined the company.

This is why some of us may be pardoned for thinking that the article "Food for Thought" in the May issue of the JOURNAL was perhaps misdirected. Few will disagree with the sound commonsense contained in Mr. Cleaver's criticism of restrictive and foolish attitudes. But we suggest that "efficient use of present staff" by management is a factor of primary significance. Individual effort is, so to speak, an *n*th-order term which, although not to be neglected, is certainly overshadowed by the effect of the employer's responsibility.

The letter in the July issue, in our opinion, misses the point. Shorter hours and better pay are not the real measure of any job. The real measure is one of satisfaction from one's work. Present-day graduates and students may be short on hours, but many are long on frustration and disillusion.—K.T.O.

SUMMER PARTY

STUDENTS AND GRADUATES turned out in force for the summer party at 4 Hamilton Place on 22nd June. More than 160 were there and, owing to space limitations, the selling of tickets had to be halted over a week before the party. The demand was great; nearly fifty postal applications for tickets had to be returned.

Among those who came to the party were the President, Mr. E. T. Jones, with his wife and daughter, and the Secretary, Dr. A. M. Ballantyne.

A highlight of the party was the team egg-and-spoon race, the final stage of which was run with competitors on all fours holding the spoons in their mouths. Raw eggs eliminated any chance of cheating; there was no hope of replacing an egg surreptitiously after it had fallen out of the spoon.

Success of the party was largely due to the committee members who laboured to decorate 4 Hamilton Place, and who made sandwiches and served behind the bar, where connoisseurs could choose from a wide selection of wines. Members of the Society's staff were very helpful; without their aid the party could not have been held.

We hope that Students and Graduates who were unable to come will be at the Section's annual dance, which is to be held at the end of November.—J.R.C.

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Reviews

CREEP AND FRACTURE OF METALS AT HIGH TEMPERATURES. *Proceedings of a Symposium held at the National Physical Laboratory in 1954. Her Majesty's Stationery Office, 1956. 419 pp. Illustrated. 30s.*

For a period measured in millennia, metallurgy was a craft. Then from the Dark Ages onwards, it began to change into a Science. Great names are rightly associated with this metamorphosis, so potent in its influence on material civilisation. To mention but a few, as a prelude to surveying the latest phase—some four centuries ago, Leonardo da Vinci carried out tensile tests. Soon after our Civil War Hooke discovered the elasticity of metals. His law was, and is of great importance although, with ever-increasing precision of measurement, it may well be found to be a practical convenience rather than an eternal verity. Shear, knowing no parents, is something of a foundling. As the mechanics of beams became understood, its existence was realised, but the determination of the pin shear ultimate of metals, a property surely as important as their ultimate tensile, has probably only been the subject of testing for about a century. Fatigue, hazily conceived before the French Revolution, was not however clearly recognised until early in the 19th century. Credit for this is due both to Poncelet and to the almost unknown German engineer, Albert. Creep, metallurgy's current and to date most difficult problem child, has been within our consciousness for but little over half a century.

It is probably right to attribute its discovery to Phillips, but it is to Andrade, who occupies a position in this field akin to that of Gough with regard to fatigue, that by far the greatest credit is due. For the past fifty years at ever-increasing tempo, inspired by the ever-increasing temperature of engineering and chemical operations, great efforts have been made to elucidate this most complex phenomenon. Under constant stress, probably even when this is below the engineers' elastic limit, the behaviour of metals, of which in elemental form there are over 70 (and more than 4,000 as commercial alloys), is greatly influenced by many variables. Among the many, one may cite the few: atomic patterns, of which there are 14 varieties; grain size, which may range from thousandths of an inch to inches; metallographic structure, which can have a diversity as great in form as that of roses in colour; phase stability, which may be as unvarying as the facial expression of the Sphinx or as transient as the pattern of the kaleidoscope, and last and most certainly not least, temperature, which can range from absolute zero to near to the melting point of the metal. Creep is indeed a phenomenon of vast complexity, at present by no means completely understood.

The proceedings of the symposium held at the National Physical Laboratory constitute a contribution to knowledge dimensionally larger than a drop in the ocean but smaller than a good thunderstorm. Over twenty papers were presented. They were divided into four classes: Deformation Processes in Simple Materials; Creep Resistance of Complex Materials; Theory of Fracture; Work on Tertiary Creep and Fracture. The merit of the various contributions is of course unequal, but as one would expect under such auspices, no paper is bad, although one, mercifully short, seems unnecessary. Many are good and a few better than this. The volume is without a satisfying review as to what is at present known and what is unknown. As with all symposia, its structure has much in common with a patchwork quilt. The components largely lack both order and a third dimension—depth is hard to find. An active worker in this field of metallurgical investigation should, nevertheless, read these papers. They will surely add, though somewhat disjointedly, to his knowledge. On the other hand, anyone wanting to obtain a broad understanding of creep, would do better to study one of the several books on this difficult subject. These have the advantage of telling a straight tale straightly. Further, since they will doubtless contain a subject index, they possess a substantial virtue not to be found in the work under review.

Having read very many publications akin and often very inferior to the present one, a generalisation may be allowed. The leading participants in a symposium seem hugely to enjoy themselves. Some, as is evidenced by the printed record of their copious utterances even become intoxicated by the exuberance of their own verbosity. They apparently attain a state of ecstasy beyond that producible by the most potent of contemporary drugs. The reader of the report of these junketings, on the other hand, has no such feelings of elation. He knows he has not been offered a flask of Horace's rich Falernian wine, nor even a mug of Inglesby's small beer—a British Railways chipped cup of watery and not very warm cocoa is about the form.—P. L. TEED.

INSTRUMENT ENGINEERING, VOLUME III; APPLICATIONS OF THE INSTRUMENT ENGINEERING METHOD; PART I, MEASUREMENT SYSTEMS. *C. S. Draper, W. McKay, S. Lees. McGraw-Hill, New York, 1955. 879 pp. £6 11s. 6d.*

Reviewers of the first two in this series of three mammoth volumes on Instrument Engineering have remarked on the fantastic excesses of the "self-defining notation" of the M.I.T. Instrumentation Laboratory. Even Professor Draper found it necessary to joke about it in his 1955 Wilbur Wright lecture! The reviewer refers to this at the start because all but the most hardy spirits will be put off by the devastating effects of this notation. That said, we may enquire what will be found by those who dig among these 879 elegantly printed pages and what class of reader all this is meant for.

This Part I of Vol. III is intended to illustrate the problems of indication and measurement by "self contained unit instruments" and by systems formed of "open chains of operating components". The chapters carry on from Vol. II, numbered from 28 to 35, and their contents, as given in the book, are as follows.

Chapter 28 contains a review of the concepts and notation used by various authors to describe operating components and operating systems. Chapter 29 gives a discussion of the generalised concepts of indication and measurement. Chapter 30 is a summary, in tabular form, of information about many of the physical principles and devices that may play important roles in measurement and control equipment. Chapter 31 is an engineering treatment of temperature-measuring instruments. (But no mention of thermistors!)

Chapter 32 builds up the theory of electro-mechanical

measuring instruments to show that typical examples have second-order performance functions. Chapter 33 uses pressure measuring as the engineering problem for developing the theory of open-chain systems. Chapter 34 describes and develops the theoretical background for a number of typical integrators and differentiators and includes a theoretical discussion of analogue computors. (But this is not a patch on Wass's book!) Chapter 35 describes and discusses the theory of instruments and systems for measuring mechanical vibrations.

The book might have been produced in response to a government's demand for more and more technologists and this could have been a manual of the department of instrument engineering for the masses. Thus we have teutonic thoroughness combined with new world simplicity. No detail is left out and the reader is helped over most of the real hurdles as well as over a multitude of imaginary ones. This is typified by statements like "Nonvectors lack directional properties and are called scalar quantities", "Computation involves the operations encountered in arithmetic, algebra, trigonometry and calculus as well as the generation of other functions sometimes empirical ones", "current carrying conductors in a magnetic field are of great importance. . ." We may spurn the use of steam hammers to crack nuts but there is no shortage of steam over there, nor of really keen young men who will plough through this sort of thing because they want to learn as much as they possibly can. We may try to dismiss the present volume, but we cannot ignore the example of the hard-working American youngsters for whom it must have been written.—P. T. FINK.

ROCKETS AND GUIDED MISSILES. *John Humphries. Ernest Benn. London, 1956. 229 pp. Illustrated. 30s.*

Here we have a well-marshalled summary, presented in clear and readable style which describes much of the really startling amount of experimentation and development work done in this field since 1943.

The bias of the book is strongly towards the matter of liquid fuel motors and to the innumerable problems which have had to be solved in their many applications. On page 51, the author tells us that B.M.W. in Germany actually tested no less than six thousand "promising looking combinations" some of them almost to be described as fantastic. No wonder then that so great a proportion of this work deals with these fuels and their oxidants and with the often elaborate, and it must be admitted, temperamental mechanisms for employing them. Other means of propulsion are not altogether neglected. Indeed the comparative diagrams on pp. 114, 115 and 116 will be most clear, even to the dilettante layman.

This remark applies to all the diagrams and tables, although the plates do not always adjoin their text. The sources are, as may be imagined, predominantly German and American, in fact Mr. Humphries hints that the vast expenditure by the United Kingdom has led to no great results. He is quite blunt in asserting the continuing need for manned interceptor aircraft to make up for the shortcomings of guided missiles.

The reviewer would be glad to have heard a good deal more about the pros and cons of equipments already used in action, but the author tends rather to gallop after the red herrings of inter-stellar or inter-planetary space while disregarding the need to keep Red infantry out of Ostend. He leaves one to guess the importance of shaped charges in warhead and the reasons for not using them in the ground attack rockets of the Hurricane and Typhoon. Neither does he go into the effect of the inherent inaccuracy of single nozzle rockets and indeed on p. 115 puts the technical cart before the tactical horse. The "heavy payload" and the "high rate of fire" of "landing craft, rocket," was a penalty commanders had to pay for the great dispersion of their missiles. Nor does he expound the military disadvantages of "Honest John" and of its little "bazooka" brother. On the other hand, he omits the very real achievements of our ballistite filled air-to-air rockets in the Battle of the Somme, due to Lieutenant Le Prieur. A minor error is the statement that A. M. Low began his guided missile work in 1917. It was 1916. Again spin stabilised rockets do not necessarily have less drag than well designed unrotated missiles, as William Teli might have demonstrated.—L. V. S. BLACKER.

LET'S LEARN TO FLY. *Arnold Warren. Isaac Pitman, London, 1956. 220 pp. Illustrated. 18s.*

"This book," says the author, "is intended for the purpose of teaching the principles of flight. It endeavours to cover, with considerable attention to detail, all the things necessary to take the pupil to the first-solo stage with a sound foundation under his flying. The use of mathematics has been avoided as far as possible."

Most pilots, whether they learned to fly in the R.A.F., at a civilian club or at a gliding club, will have consulted a book for exactly this purpose. The Elementary Training Handbook of the R.A.F. covers much the same ground as Arnold Warren's book and, to my recollection, did it very competently. It is somewhat of a surprise, therefore, to find yet another elementary book on the principles of flight. To a teenager, however, a flying primer written when he was in his cradle would appear antiquated indeed.

The chapter sequence is logical, from a discussion of elementary mechanics, via lift and drag to controls, stalling, take-off and landing, and spinning, to the first solo. The style is easy, and the treatment, almost everywhere, sound. Some of the facts quoted would not be true of all aeroplanes, but the author is clearly sketching and thinking of the Chipmunk. With such an aeroplane, maximum endurance (p. 84) is not obtained at the minimum drag speed, but at a rather slower speed, the minimum power speed. There is also confusion between these two speeds on p. 57, but no great harm is done.

"The gyroscopic effect of the spinning propeller produces what is probably the greatest yawing tendency on take-off." (p. 117). This is quite wrong. The rotation of the slipstream from a single propeller causes the well-known "swing." The additional gyroscopic effect is felt only if the tail is rapidly raised or lowered during the take-off—not a recommended practice.

The last chapter is a valuable little essay "To the Flying Instructor." A good teacher must not only know his subject matter but also know how to teach it. This book will help him with both.—A. H. YATES.

THE SCIENCE AND PRACTICE OF WELDING. *A. C. Davies. Cambridge University Press. 1956. 502 pp. Illustrated. 18s.*

This is the Fourth Edition of a well-known practical textbook for students preparing for the City and Guilds examination in Welding Science and Technology. Since 1941, when it first appeared, there have been three new editions and two reprintings, which is a good indication of its usefulness as a textbook and as a reference book for practical welders. Considerable additions have been made in this edition to cover the many new processes of welding and cutting which have appeared in recent years.

STRENGTH OF MATERIALS. *Second Edition. Bevis Brunel Low. Longmans, Green and Co., London, 1955. 320 pp. Illustrated. 18s. 6d.*

The first edition of this text book was published in 1949 and reviewed on page 484 of Volume 54 of the Society's JOURNAL in 1950.

The remarks made at that time still apply to the first 16 of this second edition's 18 chapters. The two new chapters deal with, or rather give an introduction to, the testing of materials and the mechanical properties of materials and, for beginners, these chapters are probably adequate.

There is some surprising emphasis in the new work. For example the treatment of all types of electrical resistance strain gauges is quite detailed while a more basic instrument, the mechanical extensometer, is represented by only one example. The dust cover tells the reader that in this new edition titanium alloys have not been overlooked. This refers to some page and three-quarters of text describing the expense, difficulties and problems of obtaining and working titanium and its alloys but describes so few of its really interesting properties that the reader may well be left asking "why all the fuss?"

The original review remarked that the book was inexpensive. The present edition is some 6s. dearer but the book as a whole is still not expensive.—A.J.B.

Additions to the Library

Airlines Electronic Engineering Committee. GUIDANCE FOR DESIGNERS OF AIRCRAFT ELECTRIC INSTALLATIONS. Aeronautical Radio Inc. 1955.

Airlines Electronic Engineering Committee. GUIDANCE FOR DESIGNERS OF AIRBORNE ELECTRONIC EQUIPMENT. Aeronautical Radio Inc. 1955.

A.S.T.M. SYMPOSIUM ON METALLIC MATERIALS FOR SERVICE AT TEMPERATURES ABOVE 1600 F. (STP 174). A.S.T.M. 1956.

Aviation Facilities Study Group. AVIATION FACILITIES. Director, U.S. Bureau of the Budget. 1955.

Beauregard, N. H. (*Comp.*). ILLUSTRATIONS OF COL. LINDBERGH'S DECORATIONS, ETC. Missouri Historical Society. 1935.

Biot, M. A. THERMOELASTICITY AND IRREVERSIBLE THERMODYNAMICS (C.A.L. REP. SA-987-S-1). Cornell Aeronautical Lab. 1954.

Broglio, L. LA RICERCA AERONAUTICA E I SUOI RIFLESSI SUL PROGESSO DELL'AVIAZIONE CIVILE. Centro per lo sviluppo dei trasporti aerei. 1956.

Campbell, I. E. (ed.). HIGH TEMPERATURE TECHNOLOGY. Wiley. 1956.

Chapman, S. DISCHARGE OF CORONA CURRENT FROM POINTS ON AIRCRAFT OR ON THE GROUND (REPORT C.A.L. 66). Cornell Aeronautical Lab. 1955.

Cohen, R. PUBLIC RELATIONS IN CIVIL AVIATION. Centro per lo sviluppo dei trasporti aerei. 1956.

Corbellini, G. AFFINITA TECNICHE ED ECONOMICHE DEI TRASPORTI. Centro per lo sviluppo dei trasporti aerei. 1956.

Crawley, A. ESCAPE FROM GERMANY: A HISTORY OF R.A.F. ESCAPES DURING THE WAR. Collins. 1956.

Dryden, H. L. and others. HYDRODYNAMICS. Dover. 1956.

— L'ECOLE NATIONALE SUPERIEURE DE L'AERONAUTIQUE, 1956. Librairie Vuibert. 1956.

Einstein, A. THE THEORY OF THE BROWNIAN MOVEMENT. Dover. 1956.

Geiger, H. ALPINE PILOT. Cassell. 1956.

Hertz, H. THE PRINCIPLES OF MECHANICS PRESENTED IN A NEW FORM. Dover. 1956.

I.A.S. ENGINEERING MANAGEMENT PHILOSOPHIES (A Symposium) S.M.F Fund Paper F.F.14. I.A.S. 1956.

Kuhn, P. STRESSES IN AIRCRAFT AND SHELL STRUCTURES. McGraw Hill. 1956.

Livesley, R. K. and Chandler, D.B. STABILITY FUNCTIONS FOR STRUCTURAL FRAMEWORKS. Manchester U.P. 1956.

Lowy, R. G. A TWO-DIMENSIONAL APPROXIMATION TO THE UNSTEADY AERODYNAMICS OF ROTARY WINGS C.A.L. Rep. 75). Cornell Aeronautical Lab. 1955.

MacDonald, R.S. AN EXPANDER-COMPRESSOR METHOD OF AIR COOLING (C.A.L. project report HG-591-S-2). Cornell Aeronautical Lab. 1954.

Macmillan, R. H. AUTOMATION: FRIEND OR FOE? C.U.P. 1956.

Meteorological Office. HANDBOOK OF METEOROLOGICAL INSTRUMENTS. PART 1. INSTRUMENTS FOR SURFACE OBSERVATIONS. H.M.S.O. 1956.

Miles, J. W. UNSTEADY SUPERSONIC FLOW. Air Research and Development Command. 1955.

Ministry of Education. THE ORGANIZATION OF TECHNICAL COLLEGES (Circular 305). H.M.S.O. 1956.

Ministry of Transport and Civil Aviation. THE PLANNING OF AIR-STATIONS FOR SINGLE-ENGINED HELICOPTERS. H.M.S.O. 1956.

Moncany, L. LE BOURGET: UNE PAGE D'HISTOIRE DE L'AVIATION MARCHANDE. Club du Beau Livre de France. 1956.

Myklestad, N.O. FUNDAMENTALS OF VIBRATION ANALYSIS. McGraw Hill. 1956.

*N.A.C.A. FORTIETH ANNUAL REPORT. U.S. G.P.O. 1956.

National Physical Laboratory. PHOTOMETRY OF TELESCOPES AND BINOCULARS. H.M.S.O. 1956.

*Oppermann, A. AERONAUTICAL ENGLISH (GERMAN-ENGLISH DICTIONARY AND HANDBOOK). Verlags- und Luftreklamegesellschaft. 1955.

*— PENGUIN ATLAS. Penguin. 1956.

S.A.E. S.A.E. AERONAUTICAL DRAFTING MANUAL. 6th Edition. S.A.E. 1956.

Spaight, J. M. THE BATTLE OF BRITAIN, 1940. Bles. 1941.

Speas, R. Dixon. TECHNICAL ASPECTS OF AIR TRANSPORT MANAGEMENT. McGraw Hill. 1956.

Townsend, A. A. THE STRUCTURE OF TURBULENT SHEAR FLOW. C.U.P. 1956.

Turino, R. J. (ed.). FIFTH INTERNATIONAL AERONAUTICAL CONFERENCE. LOS ANGELES 1955. I.A.S. 1956.

Wheatcroft, S. ECONOMICS OF EUROPEAN AIR TRANSPORT. Manchester U.P. 1956.

**Items marked thus are for reference use only.*

I.A.S. Preprints

638 Surface-protection and cooling systems for high-speed flight. D. J. Masson and Carl Gazley, Jr.

639 A re-evaluation of some airframe thermal problems. Norris F. Dow.

640 Techniques and problems in testing structures at elevated temperatures. Lewis H. Abraham.

641 Operational aspects of the SAC transition to jets. Col. Richard E. Evans.

642 VTOL fixed-wing aircraft. Ken S. Coward.

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646 STOL-VTOL aircraft in the Air Mobile Army. Lt. Col. M. J. Strok.

647 Air Force logistics—some recent developments. Allen R. Ferguson.

648 Passenger airline economics. Lloyd B. Aschenbeck.

649 Flight application of target-simulator principles. William M. Kauffman.

Reports

AERODYNAMICS

Boundary Layer

See also Internal Flow—Testing and Instruments

Tests of humidity effects on flow in a wind tunnel at Mach numbers between 2·48 *and* 4. *R. J. Monaghan. C.P.* 247 (1956).

Static and Pitot pressure distributions were measured in the working section of a 5 in.×5 in. supersonic wind tunnel at nominal Mach numbers of 2·48, 3·25 and 4, over ranges of absolute humidity at the inlet from 5×10^{-5} to 3×10^{-3}.—(1.1.3).

Boundary-layer-induced noise in the interior of aircraft. H. S. Ribner. UTIA. Report 37 (*April* 1956).

At high speeds the turbulent boundary layer excites appreciable skin vibration, promoting strong noise in the interior. The convected turbulent pressure distribution can be represented as a pattern of moving waves. A running ripple in the skin follows underneath each wave; multiple reflections at the frames and stringers promote standing waves. The acoustical effects of the running ripples are calculated for an infinite sheet; this is the main result of the paper.—(1.1.0×33.1.2).

*On the transmission through a fuselage wall of boundary layer noise. G. M. Corcos and H. W. Liepmann. Douglas Report No. SM-*19570 (*December* 1955).

The report deals in preliminary fashion with the transmission through a fuselage of random noise generated on the fuselage skin by a turbulent boundary layer. The concept of attenuation is abandoned and instead the problem is formulated as a sequence of two linear couplings; the turbulent boundary layer fluctuations excite the fuselage skin in lateral vibrations and the skin vibrations induce sound inside the fuselage. The techniques used are those required to determine the response of linear systems to random forcing functions of several variables.—(1.1.3.1).

Exploratory investigation of boundary-layer transition on a hollow cylinder at a Mach number of 6·9. *M. H. Bertram. N.A.C.A. T.N.* 3546 (*May* 1956).

The Reynolds number for transition on the outside of a hollow cylinder with heat transfer from the boundary layer to the wall has been investigated at a Mach number of 6·9 in the Langley 11-inch hypersonic tunnel.—(1.1.2.4).

Investigation of boundary-layer transition on 10° *cone in Langley* 4- *by* 4-*foot supersonic pressure tunnel at Mach numbers of* 1·41, 1·61, *and* 2·01. *A. R. Sinclair and K. R. Czarnecki. N.A.C.A. T.N.* 3648 (*May* 1956).

An investigation has been made to determine the transition Reynolds numbers on a 10° cone in the Langley 4- by 4-foot supersonic pressure tunnel at Mach numbers of 1·41, 1·61, and 2·01 and over a Reynolds number range from about $0\cdot8\times10^6$ to $9\cdot5\times10^6$ per foot.—(1.1.2.4).

Compressible Flow

See also Wings and Aerofoils

An experimental investigation of a sonic jet directed upstream against a uniform supersonic flow. G. A. Watts. UTIA. Technical Note 7 (*January* 1956).

The flow configuration generated by directing a sonic jet of gas upstream into a uniform supersonic flow is shown to consist of a bow shock wave in the main flow, followed by an interface between the two flows which preceded a nearly normal curved shock crossing the jet.—(1.2.3.2).

Numerical integration methods for supersonic wings in steady and oscillatory motion. B. Etkin. UTIA. Report No. 36 (*November* 1955).

Numerical integration procedures are derived for the calculation of potential and pressure distribution on supersonic wings. For wings in steady flow the method applies to supersonic leading edges, and by an extension of Evvard's principle, to subsonic leading edges. It is applicable to thin lifting surfaces with arbitrary twist and camber, or to non-lifting wings of symmetrical section. For oscillating wings, the method is developed for supersonic leading edges at all frequencies and for subsonic leading edges at low frequencies. Arbitrary elastic modes are covered by the treatment.—(1.2.3.1×1.10.1.2).

*Application of drag-reduction methods to supersonic biplanes. M. E. Graham. Douglas Report SM-*19258 (*September* 1955).

The pressure drag of certain symmetric biplane arrangements in supersonic flow is investigated. The basic configuration is that of a Sears-Haack fuselage combined with two symmetrically located elliptic plan form wings.—(1.2.3.1).

*Optimum thickness and lift distribution for a fuselage in the presence of a prescribed wing system. M. E. Graham. Douglas Report SM-*19615 (*March* 1956).

A drag formula is utilised to obtain a generalised supersonic "area rule" expressed in terms of sources, lift and side force singularities. Simple examples of optimum fuselage source distributions and optimum fuselage lift distribution are presented.—(1.2.3.1).

A unified two-dimensional approach to the calculation of three-dimensional hypersonic flows, with application to bodies of revolution. A. J. Eggers and R. C. Savin. N.A.C.A. Report 1249 (1955).

A procedure for calculating three-dimensional steady and non-steady supersonic flows with the method of characteristics is developed and discussed. An approximate method is deduced from the characteristics method and shown to be of practical value at high supersonic speeds.—(1.2.3.1).

Calculations of the rate of thermal dissociation of air behind normal shock waves at Mach numbers of 10, 12, *and* 14. *G. P. Wood. N.A.C.A. T.N.* 3634 (*April* 1956).

Calculations of the rate of dissociation of air have been based on available theory and available experimental results with other gases.—(1.2.3.2).

Theoretical pressure distributions for some slender wing-body combinations at zero lift. P. F. Byrd. N.A.C.A. T.N. 3674 (*April* 1956).

Theoretical calculations are made of the pressure distributions for some slender, symmetrical wing-body combinations in subsonic and supersonic flow. The combinations consist first of non-lifting, swept-back wings mounted on a circular cylinder and second of such wings mounted on a body indented so that the local cross-sectional area of the combination is constant.—(1.2.3.1×1.2.1.1×1.10.1.2).

Control Surfaces

Flight investigation of the effectiveness of an automatic aileron trim control device for personal airplanes. W. H. Phillips et al. N.A.C.A. T.N. 3637 (*April* 1956).

An analytical and flight investigation has been made to determine the effectiveness of an automatic aileron trim control device in conjunction with preloaded aileron centring springs in improving spiral stability characteristics of a personal aeroplane.—(1.3.1×1.8.1.1).

Investigation by the transonic-bump method of a 35° *sweptback semispan model equipped with a flap operated by a series of servovanes located ahead of and geared to the flap. W. H. Phillips and R. F. Thompson. N.A.C.A. T.N.* 3689 (*April* 1956).

Lift, drag, pitching-moment, rolling-moment, and yawing-moment data in the Mach number range from 0·6 to 1·0 obtained from wind-tunnel tests of low aspect ratio swept-back aerofoil model with a servo-vane control are presented. The control utilises the drag force and spoiler action of a set of vanes to deflect a flap-type control.—(1.3.4×1.10.2.2).

NOTE.—*The figures in parenthesis at the end of each Summary are for office use only.*

FLUID DYNAMICS ("CLASSICAL AERODYNAMICS")

Determination of vortex paths by series expansion technique with application to cruciform wings. A. Y. Alksne. N.A.C.A. T.N. 3670 (*April* 1956).

A series method is applied to the computation of vortex positions behind a slender equal-span cruciform wing at any angle of bank as a function of the distance behind the trailing edge.—(1.4.3 × 1.6.1).

INTERNAL FLOW

Alternating pressures and blade stresses in an axial-flow compressor. J. R. Forshaw, et al. R. & M. 2846 (1956).

An investigation has been made into the origin of the forces exciting vibration in an axial-flow compressor, and their magnitude relative to the calculated gas bending loads. The blade stresses resulting from these forces were measured, and a value obtained for the energy input to a blade when vibrating in the predominant modes under running conditions. The experimental data were obtained from pressure elements and strain-gauges fitted to the inlet guide vanes and first four stator-blade rows of an early compressor.—(1.5.2.1 × 1.6.3).

On the gas dynamics of a rotating impeller. A. Busemann. N.A.C.A. T.M. 1385 (*March* 1956).

It is shown that, for a compressible flow with constant entropy, the pressure rise maintains the direct relation to the circulation around the blades existing for incompressible flow. In contrast, however, the torque, and with it the power consumption, is increased because of sound waves travelling to infinity already at subsonic circumferential speeds.—(1.5.3.1).

Discharge coefficients for combustor-liner air-entry holes. 1—Circular holes with parallel flow. R. T. Dittrich and C. C. Graves. N.A.C.A. T.N. 3663 (*April* 1956).

An experimental investigation was conducted to determine the effects of hole diameter, wall thickness at hole, duct height, parallel flow velocity, pressure ratio, boundary layer thickness, and static pressure level on the discharge coefficient for circular holes having flow parallel to the plane of the hole.—(1.5.1.3).

Perforated sheets as a porous material for distributed suction and injection. R. E. Dannenberg, et al. N.A.C.A. T.N. 3669 (*April* 1956).

Measurements were made of the resistance to air flow through a series of perforated metal sheets with open areas ranging from less than 1 per cent. up to 41 per cent. The use of punch-perforated sheets as a porous material for boundary layer control applications by means of distributed suction is discussed.—(1.5.1.3 × 1.1.5.1).

Some effects of guide-vane turning and stators on the rotating stall characteristics of a high hub-tip ratio single-stage compressor. E. L. Costilow and M. C. Huppert. N.A.C.A. T.N. 3711 (*April* 1956).

The rotating stall characteristics of several guide-vane-rotor and guide-vane-rotor-stator configurations were investigated. Guide vanes having turning angles of −22·5°, 0°, 22·5°, and 40° were used. The stage hub-tip ratio was 0·9.—(1.5.2.1).

LOADS

See also FLUID DYNAMICS—INTERNAL FLOW

Generalized indicial forces on deforming rectangular wings in supersonic flight. H. Lomax, et al. N.A.C.A. Report 1230 (1955).

A method is presented for determining the time-dependent flow over a rectangular wing moving with a supersonic forward speed and undergoing small vertical distortions expressible as polynomials involving spanwise and chordwise distances. Results are expressed in terms of generalised indicial forces.—(1.6.3).

Analysis of a vane-controlled gust-alleviation system. R. W. Boucher and C. C. Kraft. N.A.C.A. T.N. 3597 (*April* 1956).

A theoretical study, using an electronic analogue computor, is made of the response to step gusts of an aeroplane equipped with a system in which modified wing flaps and a modified elevator are operated to reduce accelerations in rough air. These surfaces are actuated by an automatic control system in response to the indications of an angle-of-attack vane. The effect of interconnection of the flap operating mechanism with the pilot's control system is also included.—(1.6.3).

Initial results of a flight investigation of a gust-alleviation system. C. C. Kraft. N.A.C.A. T.N. 3612 (*April* 1956).

A flight investigation of a gust-alleviation system designed to improve passenger comfort in flight through turbulent air has been recently initiated. The system is designed to maintain constant lift in rough air by means of trailing edge wing flaps operated by an automatic control system which is controlled by signals from an angle-of-attack vane located at the nose of the aeroplane. The gust-alleviation system is at least capable of reducing the normal accelerations due to gusts by 50 per cent. in the range of the natural frequency of the aeroplane.—(1.6.3).

STABILITY AND CONTROL

See also WINGS AND AEROFOILS—CONTROL SURFACES

A flight evaluation of the longitudinal stability characteristics associated with the pitch-up of a swept-wing airplane in maneuvering flight at transonic speeds. S. B. Anderson and R. S. Bray. N.A.C.A. Report 1237 (1955).

Flight measurements on a swept-wing jet aircraft showed that the pitch-up encountered in a wind-up turn at transonic Mach numbers was due principally to an unstable break in the wing pitching moment associated with flow separation near the wing tip. The pitch-up encountered in slowing down in a dive-recovery manœuvre was due chiefly to a reduction in wing-fuselage stability.—(1.8.2.2 × 13.2).

Theoretical and analog studies of the effects of nonlinear stability derivatives on the longitudinal motions of an aircraft in response to step control deflections and to the influence of proportional automatic control. H. J. Curfman. N.A.C.A. Report 1241 (1955).

Through theoretical and analogue results the effects of two non-linear stability derivatives on the longitudinal motions of an aircraft have been investigated. Non-linear functions of pitching-moment and lift coefficients with angle of attack were considered. Analogue results of aircraft motions in response to step elevator deflections and to the action of the proportional control systems are presented.—(1.8.2.1).

Some effects of fuselage flexibility on longitudinal stability and control. B. B. Klawans and H. I. Johnson. N.A.C.A. T.N. 3543 (*April* 1956).

Some effects of fuselage flexibility on longitudinal stability and control of a large high-speed aeroplane were determined by application of a semi-rigid analysis. The dynamic characteristics are examined by studying the period and damping of the fuselage and aeroplane short-period longitudinal modes of oscillation. The static characteristics are examined by studying the longitudinal stability margins and the elevator-control deflections required for balance in straight or steady manœuvring flight.—(1.8.2.1 × 2).

Analysis of a spin and recovery from time histories of attitudes and velocities as determined for a dynamic model of a contemporary fighter airplane in the free-spinning tunnel. S. H. Scher. N.A.C.A. T.N. 3611 (*April* 1956).

A technique for determining time histories of the attitudes and velocities of free-spinning-tunnel models during spins and recoveries is presented and applied to a model representative of a contemporary fighter aeroplane. One spin and the recovery therefrom are investigated, and discussions as to the nature of the spin and recovery motions and some factors affecting them are presented along with the time histories.—(1.8.3.1).

WINGS AND AEROFOILS

See also COMPRESSIBLE FLOW—CONTROL SURFACES

An investigation of the maximum lift of wings at supersonic speeds. J. J. Gallagher and J. N. Mueller. N.A.C.A. Report 1227 (1955).

Results of an investigation are presented for a variety of wing plan forms of random thickness distributions tested at Mach numbers of 1·55, 1·90, and 2·32 and at angles of attack ranging from zero up through the angle at which maximum lift occurred.—(1.10.2.2×1.2.3.2).

An analysis of estimated and experimental transonic downwash characteristics as affected by plan form and thickness for wing and wing-fuselage configurations. J. Weil, et al. N.A.C.A. T.N. 3628 (*April* 1956).

A summary is presented of the effects of changes in wing plan form and thickness ratio on the downwash characteristics of wing and wing-fuselage configurations in the Mach number range between 0·6 and 1·1. Data obtained by the transonic-bump technique at two tail heights has been compared with theoretical estimations made in the subsonic and supersonic Mach number range.—(1.10.2.2×1.8.2.1).

Investigation at high subsonic speeds of a body-contouring method for alleviating the adverse interference at the root of a sweptback wing. J. B. McDevitt and W. M. Haire. N.A.C.A. T.N. 3672 (*April* 1956).

An experimental investigation was made of a body-contouring method for alleviating at subsonic speeds the adverse interference at the root of a 35° swept-back wing in combination with various bodies.—(1.10.2.2×1.2.2.1).

Helicopter Aerodynamics

Normal component of induced velocity in the vicinity of a lifting rotor with a nonuniform disk loading. H. H. Heyson and S. Katzoff. N.A.C.A. T.N. 3690 (*April* 1956).

Part I presents a method for computing, from available calculations for uniform disc loading, the effect of non-uniform circularly symmetrical disc loading on the normal component of induced velocity in the vicinity of a lifting rotor.—(1.11.3).

Analysis and comparison with theory of flow-field measurements near a lifting rotor in the Langley full-scale tunnel. H. H. Heyson. N.A.C.A. T.N. 3691 (*April* 1956).

Measurements of stream angles and velocities were made in several planes both near and behind a single rotor in four different flight conditions in the Langley full-scale tunnel.—(1.11.3).

Testing and Instruments

See also Flight Testing

Experiments with slotted and perforated walls in a two-dimensional high-speed tunnel. D. W. Holder, et al. R. & M. 2955 (1956).

Preliminary tests have been made with a 7½ in.×3 in. two-dimensional induced-flow tunnel fitted with slotted or with perforated walls, and with a closed tunnel and an open jet of the same size. The experiments include observations of the pressures in the empty tunnel, and of the pressure distribution and flow pattern round a 2 in. chord aerofoil at angles of incidence of 0° and 2°.—(1.12.1.2).

A study of the china-film technique for flow indication. R. J. Stalker. ARL. Report A.96 (October 1955).

Experiments on the mechanism of formation and the aerodynamic meaning of China-clay streak patterns are reported. Their formation is shown to be intimately connected with a sub-layer of sludge in the liquid film, and the definition of the pattern is shown to depend on the viscosity of the carrier liquid.—(1.12.1.3×1.12.1.1×1.1.2.4×1.1.2.1).

Wind-tunnel investigation of a number of total-pressure tubes at high angles of attack: subsonic, transonic, and supersonic speeds. W. Gracey. N.A.C.A. T.N. 3641 (*May* 1956).

The total-pressure errors of 54 total-pressure tubes at angles of attack up to 60° and Mach numbers from 0·26 to 1·62 are presented. The configurations of the various tubes differed as regards external shape, internal shape, and type of total-pressure entry.—(1.12.5).

AEROELASTICITY

See also Aerodynamics—Stability and Control

The flexure-torsion flutter of cambered aerofoils in cascade. A. H. Craven and I. Davidson. CoA. Report 95.

Results are given of a series of tests on the flexure-torsion flutter of cascades of aerofoils of 30° and 45° camber. The critical flutter speeds and frequencies in cascade are expressed as ratios of the values for the aerofoil in the isolated condition. The tests cover stagger angles between −30° and +30° and gap chord ratios up to 1·5 at a Reynolds number of 1×10^5 based on aerofoil chord.—(2).

On panel flutter and divergence of infinitely long unstiffened and ring-stiffened thin-walled circular cylinders. R. W. Leonard and J. M. Hedgepeth. N.A.C.A. T.N. 3638 (*April* 1956).

A preliminary theoretical investigation of the panel flutter and divergence of infinitely long, unstiffened and ring-stiffened thin-walled circular cylinders is described. Analytical criteria for the location of stability boundaries are presented along with a limited number of computed results.—(2).

ELECTRONICS

On the synthesis of 3-terminal RC networks. K. M. Adams. CoA. Report 96.

This report concerns the synthesis of 3-terminal RC networks according to three specified network functions. Some necessary conditions for physical reliability are derived from first principles by an algebraic method. The possibility of synthesis is shown to depend upon a certain inequality but the weakest conditions required in order to satisfy it are not yet known.—(11).

FLIGHT TESTING

See also Aerodynamics—Stability and Control

The dynamic-response characteristics of a 35° swept-wing airplane as determined from flight measurements. W. C. Triplett, et al. N.A.C.A. Report 1250 (1955).

Longitudinal and lateral directional dynamic response characteristics of a 35° swept-wing fighter aeroplane are presented. Frequency responses, transfer functions, and stability derivatives are evaluated from flight measurements of aeroplane responses to abrupt elevator, rudder, and aileron motions through the Mach number range of 0·50 to 1·04.—(13.2).

Results of a flight investigation to determine the zero-lift drag characteristics of a 60° delta wing with NACA 65-006 airfoil section and various double-wedge sections at Mach numbers from 0·7 to 1·6. C. J. Welsh. N.A.C.A. T.N. 3650 (*April* 1956).

Results of an exploratory free-flight investigation at zero lift of several rocket-powered drag-research models equipped with 60° swept-back delta wings are presented for a Mach number range from about 0·70 to 1·60.—(13.3×1.12.2).

HYDRODYNAMICS

Effect of shallow water on the hydrodynamic characteristics of a flat-bottom planing surface. K. W. Christopher. N.A.C.A. T.N. 3642 (*April* 1956).

The effects on the planing characteristics of the clearance between a flat-bottomed planing surface and the tank bottom are presented.—(17.2).

MATERIALS

See also Science—General. Structures—Theory and Analysis

A review of plastics for use in the aircraft industry. F. Fahy. CoA. Note 43 (1956).

The types of plastic materials available are numerous and it was in order to select those of interest of British manufacture and to collect together their typical properties that this review was undertaken. For the purpose of this review the materials were grouped as follows:—I. Structural plastics—glass laminates, asbestos laminates and paper and cloth laminates. II. Thermoplastics—Polymethyl Methacrylate (perspex), Nylon and Polytetrafluoroethylene (fluon). III. Adhesives and IV. Foamed plastics.—(21.3.3).

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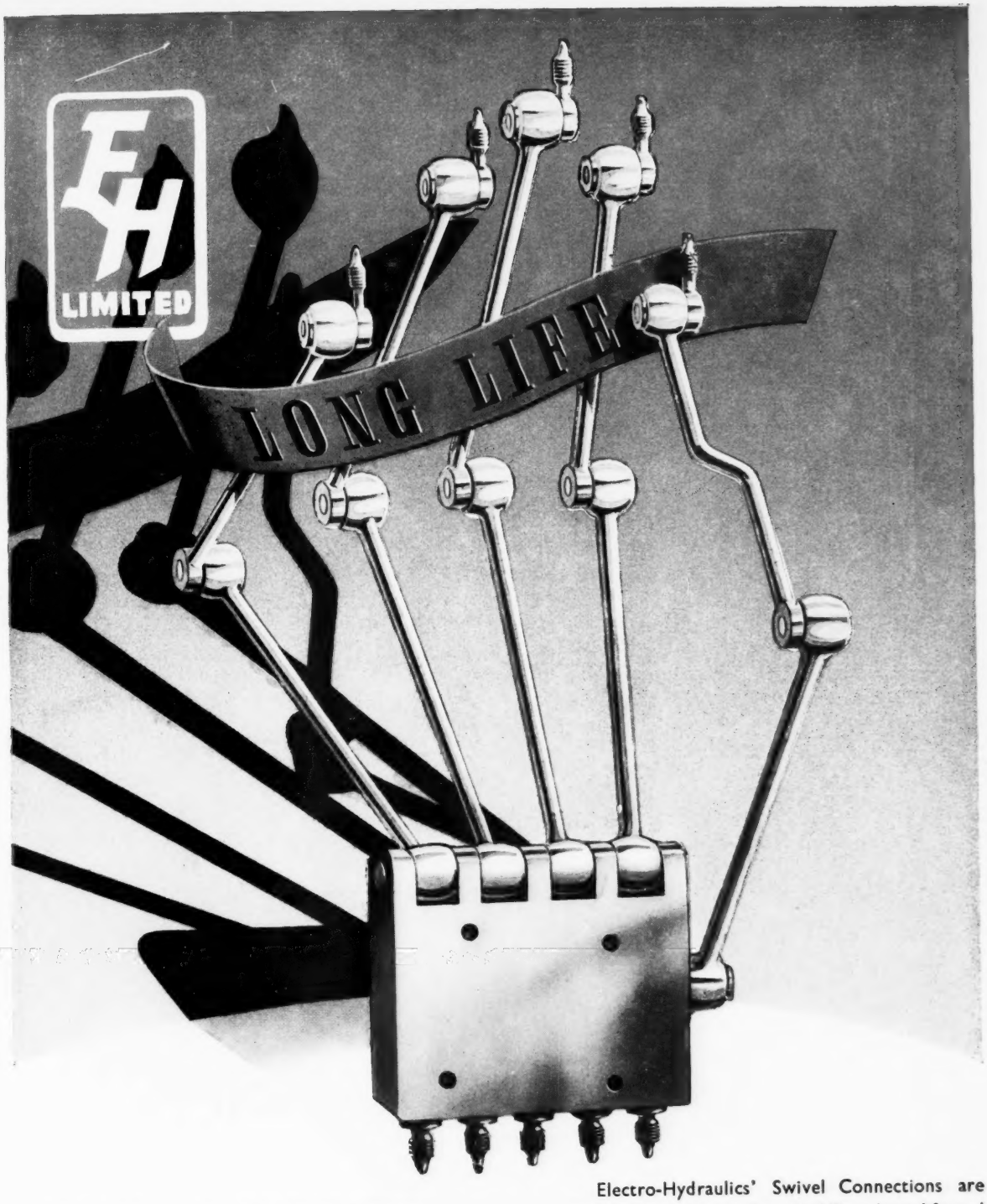

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